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DEPARTMENT OF THE AIR FORCE

AIR UNIVERSITY

AIR FORCE INSTITUTE OF TECHNOLOGY

Wright-Patterson Air Force Base, Ohio

AFIT/GSE/AA/88D-2

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PRELIMINARY DESIGN OF A MODULAR UNMANNED

RESEARCH VEHICLE

VOLUME TWO: SUBSYSTEM TECHNICAL DEVELOPMENT

DESIGN STUDY

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PRELIMINARY DESIGN OF A MODULAR UNMANNED RESEARCH VEHICLE

VOLUME TWO: SUBSYSTEM TECHNICAL DEVELOPMENT

DESIGN STUDY

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We undertook the design of a Modular Unmanned Research Vehicle after we were presented with the basic idea by Major Lanson Hudson from the Department of Aeronautics and Astronautics at the Air Force Institute of Technology. This document describes the approach, analysis, and results of the design study. The document is divided into three volumes: Volume One is the System Design Document and contains the system level information; Volume Two, Subsystem Technical Development, details the design of the various subsystems which make up the MURV; and Volume Three contains the appendices.

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List of Symbols

Symbol	Description
a	Acoustic speed
A	Area
A_0	Inlet freestream area
A_1	Inlet throat area
A_{hl}	Inlet capture area of highlight
A_{BLD}	Boundary layer diverter projected area
A^*	Area corresponding to Mach flow
AR	Aspect ratio
b	Wing span
c	Wing chord
$\bar{c}$	Mean aerodynamic chord
C_D	Drag coefficient
C_{D_A}	Inlet corrected additive drag coefficient
$C_{D_{add}}$	Inlet additive drag coefficient
$C_{D_{app}}$	Drag coefficient at landing approach
$C_{D_{BLD}}$	Boundary layer diverter drag coefficient
C_H	Hinge moment coefficient
C_L	Lift coefficient
C_{L_α}	Slope of C_L versus α
$C_{L_{max}}$	Maximum lift coefficient
C_{L_0}	Lift coefficient for level unaccelerated flight
C_M	Moment coefficient
C_{M_α}	Slope of C_M versus α

C_{M_0}	C_M at $\alpha_{L=0}$
CS_e	Engine installation cross section
D	Diameter of engine face or inlet duct exit
D	Drag
D_{hl}	Inlet highlight diameter or height
DP	Differential pressure
F_g	Net or installed thrust
F_n	Net thrust
g	Acceleration due to gravity
h	Altitude
H	Hinge moment
l	Characteristic length
K_{add}	Empirical correction for lip suction
L	Lift
L	Inlet length
L_B	Boundary layer diverter offset length
L_e	Engine length
m	Mass
$\dot{m}$	Mass flow
M	Mach number
M	Moment
n	Number of bits per digital word
p	Roll rate
p, p_∞	Static pressure
p_M	Percentage of fuel at maneuver condition

p_t	Total pressure
P_t	Total pressure
$\frac{P_{t_2}}{P_{t_0}}$	Inlet pressure ratio
$\frac{P_{t_8}}{P_{t_0}}$	Nozzle pressure ratio
q	Pitch rate
q	Dynamic pressure
q_r	Number of discrete levels in digital code
r	Yaw rate
R	Universal gas constant
R/C	Rate of climb
Re	Reynolds number
s	Takeoff or landing distance
S,S_w	Wing planform area
S_c	Canard planform area
S_t	Tail planform area
t	Time
t_f	Time of fuel available
$\hat{t}$	Aerodynamic time
T	Free stream air temperature
T	Thrust
T_t	Total temperature
T/W	Thrust-to-weight ratio
u_0	Unperturbed velocity along longitudinal axis
V	True airspeed
$\mathbf{V}$	Velocity vector

$\dot{\mathbf{V}}$	Acceleration vector
V_t	Tail volume coefficient
V_∞	Freestream velocity
W_e	Engine weight
W_E	Airframe empty weight
W_F	Fuel weight
W_{PL}	Payload weight
W/S	Wing loading
x	Flat plate friction length
α	Angle of attack
$\alpha_{C_{L_{max}}}$	Angle of attack for $C_{L_{max}}$
$\alpha_{L=0}$	Angle of attack for zero lift
β	Angle of sideslip
γ	Ratio of specific heats
γ	Flight path angle
Δy	Inlet vertical offset
δ_t	Turbulent boundary layer thickness
δ_η	Control surface deflection
η	Inlet pressure recovery
η_t	Tail efficiency factor
θ_{BLD}	Boundary layer diverter compression ramp angle
Λ	Wing leading edge sweep angle
λ	Wing taper ratio
μ	Friction coefficient
μ_b	Braking coefficient

μ_r	Rolling coefficient
μ_∞	absolute viscosity coefficient
ν	Kinematic viscosity
Π_{DS}	Dynamic similarity parameter
ρ	Air density

List of Abbreviations

Abbreviation	Description
a.c.	aerodynamic center
AC	Alternating Current
A/C	Aircraft
AF	Air Force
AFB	Air Force Base
AFFTC	Air Force Flight Test Center
AFIT	Air Force Institute of Technology
AFWAL	Air Force Wright Aeronautical Laboratory
AGL	Above Ground Level
AM	Amplitude Modulation
AOA	Angle of Attack
ASD	Aeronautical Systems Division
ASL	Above Sea Level
BDR	Battle Damage Repair
c.g.	center of gravity
C.S.	Control Surface
CAM	Configuration Analysis Module (IDAS)
CDM	Configuration Development Module (IDAS)
DC	Direct Current
ECM	Electronic Counter Measures
EGT	Exhaust Gas Temperature
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
FM	Frequency Modulation

GINO	General Interactive Optimizer
HiMAT	Highly Maneuverable Aircraft Technology
Hz	Hertz (cycles per second)
IDAS	Integrated Design and Analysis System
IRIG	Inter-Range Instrumentation Group
JPG	Jefferson Proving Ground
KVA	Kilovolt-amp
lb_m	pounds mass
lb_f	pounds force
lb st	pounds static thrust
LEF	Leading Edge Flap
LRS	Launch/Recovery System
MFP	Mass Flow Parameter
MHz	Mega Hertz (10^6 cycles per second)
MOE	Measure of Effectiveness
MSL	Mean Sea Level
MURV	Modular Unmanned Research Vehicle
n.p.	neutral point
NASA	National Aeronautics and Space Administration
PAM	Pulse Amplitude Modulation
PCM	Pulse Code Modulation
PM	Phase Modulation
PSM	Parametric Synthesis Module (IDAS)
PSM	Post Stall Maneuver
PST	Post Stall Turn

RCS	Radar Cross-Section
RPM	Revolutions Per Minute
rf	radio frequency
RPV	Remotely Piloted Vehicle
SFC	Specific Fuel Consumption
SNF	Supernormal Flight
SRV	Spin Research Vehicle
TMN	Throat Mach number
TOGW	Takeoff Gross Weight
UHF	Ultra High Frequency
URV	Unmanned Research Vehicle
USAF	United States Air Force
WPAFB	Wright-Patterson Air Force Base

Abstract

This thesis presents the analysis and development of a modular unmanned research vehicle (MURV) to support aeronautical research for the Air Force Institute of Technology. The MURV is proposed as a test vehicle to permit experimental efforts beyond the restrictions of pure analytical and wind tunnel research, yet less costly and more accessible than full-scale flight tests. A classical systems approach was applied, in concert with a conventional aircraft design process, which emphasized system level needs and objectives in the design of MURV subsystems. The primary design drivers were the need for adequate data acquisition for anticipated experiments, structural and functional modularity to permit simple reconfiguration, and focus on a set of unique experiments relating to fighter-like supermaneuverability. The supermaneuverability experiments dictated that the general arrangement of the MURV baseline design would resemble a typical modern fighter aircraft configuration, the recommended baseline being a turbojet-powered delta wing design with canards, single vertical tail, and control-configured ventral fins. Modularity implications resulted in the design of a flexible, digital flight control system with primary functions distributed between the vehicle and a remote pilot/control ground station, and a fuselage design which allows for relocation and replacement of wings and tails or canards. The data acquisition system is fully integrated with the flight control system and the remote ground station. The MURV is capable of flight speeds approaching 260 knots for altitudes up to 20,000 feet, and has fuel to fly for well over 30 minutes. Several follow-on studies are identified which are necessary to complete the design and bring the MURV to an operational status.

PRELIMINARY DESIGN OF A MODULAR UNMANNED RESEARCH VEHICLE

I. Introduction

This volume contains the technical development of the MURV design. Once the free-flying vehicle concept was selected, the design of the MURV evolved to the point where subsystems were identified. For each subsystem a subset of the MURV Objective Hierarchy was identified and used to refine subsystem objectives and measures of effectiveness, see Volume One, Figure 2.2 . The systems approach was then applied to generate and select the best design for each subsystem. Chapters One through Ten describe the development of each of the subsystems from their inception. Interacting subsystems were developed by coordinating the definition of requirements and addressing the important interface considerations for each subsystem design. The resulting preliminary designs were described in Volume One, Chapter Five. Chapter Eleven describes the process that led to the selection of the preliminary design configuration, MURV-320.

II. *Vehicle Configuration Development*

This section presents the criteria and analyses which led to the preliminary definition of a vehicle sise and alternate MURV configuration options. The primary influencing factor in defining the sise for the MURV was the mission definition developed in Section 4.4 of Volume One. Among the design criteria applied in synthesising the configuration options were those applied in the Conceptual Design Phase. In Preliminary Design, a greater number of influencing factors were considered in the design of the external arrangement and subsystems, and to greater detail. A typical example is the fuselage shape. Initially designed to represent the long, slender shape of a supersonic body, it subsequently became heavily influenced by concerns for modularity and maintainability. Tradeoffs such as these became the rule for defining the external shape of the MURV.

Initial sise estimates for the vehicle had to be made before the shape could realistically be defined. The first subsection decribes how the required sizing parameters were defined for this phase. Once the size was known, the external shaping work was initiated. This effort is the subject of the second subsection.

Selection of the recommended configuration is not described here, however. Several factors which influenced this decision are discussed in other chapters of this volume, such as modularity and flight controls integration. To avoid repetition, and to demonstrate the influence of the other design disciplines on the decision criteria, the decision making process for selecting a single airframe is not discussed until all other vehicle subsystem descriptions are presented.

2.1 Vehicle Sizing

As applied here, vehicle *size* refers to the values of three specific vehicle sizing parameters: takeoff gross weight (TOGW), thrust-to-weight (T/W), and wing loading (W/S). TOGW is the most familiar parameter of the three, as it gives a direct indication of the physical size of an aircraft. Thrust-to-weight and wing loading do not indicate the absolute size of an aircraft, but they say more about its performance. Each was estimated in conjunction with design and performance constraints which evolved from the system needs and objectives.

2.1.1 Estimating TOGW Though TOGW is a relatively simple parameter to calculate and understand, it is perhaps the most difficult of all design parameters to control. Several aircraft in recent history, particularly fighters, have had significant problems with increases in gross weight, mostly unforeseen at the beginning of the development program. The F-16 Falcon had its birth in the USAF "Lightweight Fighter Program" in 1972, whose aim was to reverse the fighter design trend of heavier, more expensive airframes [43:7–20]. The early development of the USAF F-111 had weight problems so severe that a special "Super Weight Improvement Program" was needed to try to shed gross weight. When that program proved insufficient, a second "Colossal Weight Improvement Program" was then started [90:14]. The USAF F-15 Eagle went into service with a 56,000 pound gross weight limitation, and now, modified as an F-15E Strike Eagle, has a maximum gross weight of 81,000 pounds. History indicates that, whether intended or not, the weight of an aircraft grows throughout development and, as in the case of the F-15E, can grow substantially after deployment. With these sobering facts in mind, we went about the task of estimating the takeoff gross weight of the MURV, fully intent on maintaining as realistic an estimate as possible.

The gross weight of a vehicle is the sum of all its individual component weights. The component weights are often grouped into common categories, such as operating empty weight, expendable or store weights, payload weight, and fuel weight, or W_{OE}, W_{STO}, W_{PL}, and W_F, respectively [83:5–7]. Payload weight includes any equipment which is temporarily carried internally, and is not part of the basic structure or equipment of the aircraft. This would generally include items such as cargo, passengers, internally loaded bombs and missiles, and, most applicable to the MURV, experiment-unique test equipment. Store weight includes any items carried externally, such as wing-mounted or fuselage-mounted bombs and missiles, and wing-mounted or fuselage-mounted pods and pylons. Fuel weight is simply determined from the amount of fuel carried on-board. No external stores of any kind were considered in the Preliminary Design Phase. Therefore, no store weights or external fuel weights were assessed in estimating the takeoff gross weight for the MURV.

W_{OE} is further broken down into empty weight, W_E, and weight for trapped fuel and oil, W_{TFO}. W_{TFO} accounts for the fact that there is always a small amount of fuel that is trapped in the fuel tank and/or oil sunken into a drain reservoir that cannot be used, and is not normally removed during routine maintenance. Empty weight includes the basic structural weight of the vehicle and the weight of all fixed equipment; that is, equipment which stays in or on the vehicle under normal circumstances. This includes components such as the engine, control surface actuators, fuel lines, standard guidance and control equipment, antennae, and on-board computers which are part of the basic vehicle design.

These weights are related as follows:

$$TOGW = W_{OE} + W_F + W_{PL} + W_{STO} \tag{2.1}$$

where,

$$W_{OE} = W_E + W_{TFO} \tag{2.2}$$

To estimate the gross weight, an iterative procedure was used which is based on historical data of the takeoff gross weight and empty weight of many aircraft of various types, from home-built single engine airplanes to supersonic-cruise transports and bombers [83:7–48].

The method requires preliminary estimates of TOGW to converge on a solution for W_E, which is calculated in two separate ways. When the two calculated values for W_E are within an acceptable range, the final TOGW is the estimated value used in the last, converging iteration.

An example of the procedure is given below. The description of each step is a summary of the procedure given in detail in Reference [83]. This example represents the first attempt at sising the MURV according to weight estimates available at a very early stage in the development process.

1. Estimate the total payload weight, W_{PL}. In estimating the payload weight, consideration was given to the additional equipment that might be needed in performing a variety of experiments. The first estimate came from a comparison of the XBQM-106 research vehicle, operated by AFWAL/FIGL at Wright Patterson AFB. The XBQM-106 has a payload capacity of about 30 pounds, and is used only for flight control experiments [65]. Since the MURV is expected to perform a broader range of experiments, a greater payload capability was necessary. Therefore, the initial estimate of payload weight capacity was established at 50 pounds.

2. Estimate the required fuel weight, W_F. The fuel weight estimate was based on assuming the engine would operate at maximum power for thirty minutes at sea level. This did not reflect any particular experiment's test requirement, other than to specify a maximum flight duration. For most experiments, this amount of fuel would permit a flight time of considerably longer than thirty minutes since the engine would probably not operate at maximum power throughout the flight. Also, because fuel consumption decreases as altitude increases, the assumption of sea level operation caused the initial estimate to be even more conservative.

The amount of fuel required for this simplified mission depended on the fuel consumption characteristics of the selected engine. For this initial iteration, the engine had not yet been selected. One of the candidates was a prototype engine developed by Microdynamics Corporation of Indianapolis, Indiana; this engine was used in the initial weight estimate. To maintain consistency throughout the Preliminary Design Phase, the Microdynamics engine's weight, size, thrust, and fuel flow were used in all calculations of MURV sizing and performance.

The maximum gross thrust produced by the Microdyne engine is quoted at 70 pounds, with a fuel flow rate of 65 pounds per hour at sea level. Even though the engine weighs only 10 pounds, with a payload of 50 pounds plus the weight of instrumentation and structure, a reasonable thrust-to-weight was not achievable with only one engine. Therefore, a twin-engine configuration was used, resulting in a total fuel flow rate of 130 pounds per hour. Therefore, the fuel required for operating thirty minutes at sea level under full power was 65 pounds.

3. Estimate a value of gross weight, $TOGW_{est}$. The initial value for TOGW was rather arbitrary since the method converges to a solution after several iterations of TOGW estimates. For the first iteration, TOGW was estimated at 150 pounds.

4. Estimate the weight of trapped fuel and oil, W_{TFO}. Roskam suggests that this weight is often neglible in initial estimates for gross weight [83:7]. However, to maintain conservatism in the weight estimates, five pounds was allocated for trapped fuel and oil.

5. Calculate a tentative value of empty weight, $W_{E_{tent}}$. $W_{E_{tent}}$ was calculated by combining Equations 2.1 and 2.2 and solving for $W_{E_{tent}}$:

$$W_{E_{tent}} = TOGW_{est} - W_F - W_{PL} - W_{TFO}$$

Using the weights previously defined, the initial estimate of $W_{E_{tent}}$ was

$$W_{E_{tent}} = 150 - 65 - 50 - 5 = 30 \text{ (lb)}$$

This is the weight dedicated to all structure and fixed equipment in the vehicle. This value appeared to be far too low since the engines alone weighed over 20 pounds. The results of the first iteration of this procedure would determine whether this observation was at all accurate.

6. Calculate the empty weight of the vehicle, W_E. The calculated value of W_E was found from the data base provided with the method. As previously mentioned, the data consisted of TOGW correlated with W_E for several types of aircraft. To find the best estimate of empty weight for the MURV, the most applicable data base had to be selected from among fourteen different classes of aircraft types, ranging from homebuilt models to supersonic cruise aircraft. Among the various classes, none were ideally suited for the MURV application. A composite aircraft data base was somewhat applicable since the MURV structure was expected to be composed of a high percentage of composites. The scaled-fighter data base seemed appropriate also, since that is what the MURV is designed to be. With no readily available means of combining the data bases, we decided to estimate the MURV empty weight as the arithmetic average of the values for composites and scaled fighters.

The relationship between estimated gross weight and calculated empty weight for any of these aircraft types was estimated by a log-linear regression curve-fit of the data. The estimating relationship relating TOGW to $W_{E_{calc}}$ is

$$W_{E_{calc}} = \log^{-1}\left(\frac{\log TOGW_{est} - A}{B}\right) \tag{2.3}$$

where and A and B are constants for the regression curve-fit applied to the particular data base. For composite aircraft, these constants were

$$A_{comp} = 0.8222 \quad B_{comp} = 0.8050$$

and for scaled-fighters

$$A_{sf} = 0.5542 \quad B_{sf} = 0.8654$$

Substituting these constants into Equation 2.3 and combining gave the equation for the calculated empty weight of the MURV

$$W_{E_{calc}} = \frac{1}{2}\left[\log^{-1}\left(\frac{\log TOGW - 0.8222}{0.8050}\right) + \log^{-1}\left(\frac{\log TOGW - 0.5542}{0.8654}\right)\right] \tag{2.4}$$

where the first term represents half the empty weight estimate for composite aircraft and the second term is half the empty weight estimate for a scaled-fighter aircraft. For the initial TOGW estimate, the average $W_{E_{calc}}$ was found to be 61 pounds.

7. *Compare $W_{E_{tent}}$ to $W_{E_{calc}}$* Comparing this to $W_{E_{tent}}$, a percentage difference was calculated, where the difference in the estimates was normalized by the average value. For the first iteration, $W_{E_{tent}}$ was 30 pounds and $W_{E_{calc}}$ was 61 pounds. The resulting percentage difference was 68.1%. Since $W_{E_{tent}}$ was too low, another iteration was required with a higher TOGW estimate, and the procedure was repeated beginning at Step 3.

This process was repeated until the two weight estimates were within 2.0%. This occurred for the values shown in Table 2.1, which documents the initial weight estimate for the MURV. These weight estimates were used to determine the initial values of T/W and W/S at takeoff for the early stages of the configuration development.

Table 2.1. Preliminary Weight Breakdown Estimate for the MURV

Item	Symbol	Weight (lbs)
Empty	W_E	90
Payload	W_{PL}	50
Fuel	W_F	65
Trapped F/O	W_{TFO}	5
Total	TOGW	210

2.1.2 Estimating Takeoff Thrust-to-Weight, T/W The value of T/W was intentionally restricted for the Preliminary Design Phase to that resulting from designing the MURV for two Microdynamics prototype engines. This restriction was employed for the following reasons:

Engine Scarcity. There simply are not very many small-scale turbojet or turbofan engines available which are also feasible for installation into the MURV. The comparison of such engines is presented more fully in Chapter III of this volume; it is sufficient at this point to mention that most candidates were either cost-prohibitive or sise-prohibitive. The five that were found to be feasible constituted a fairly small range of thrust capability, though the differences in fuel required for thirty-minute operation at full power were significant.

Consistency. Since the preliminary gross weight estimates had been performed assuming a particular engine, it was advantageous to continue to use that engine throughout the Preliminary Design Phase *for the purposes of configuration development only.* This did not impair the evaluation of other candidate engines for use in the MURV since the vehicle configuration was developed with the intent that any of the candidate engines could be installed.

Simplicity. With several airframe options and five feasible engine options, the matrix of airframe/engine configurations was excessive for performing a truly discriminating analysis. By using only one engine design throughout this phase, a more complete analysis of the external configurations was performed. By selecting two engines for a more complete airframe/engine analysis, we were able to devote more attention to analyzing the preferred engines.

Going forward with this philosophy, the preliminary T/W used for configuration development was found from the assumptions made in the first estimate of TOGW, previously described. From Table 2.1 the initial TOGW estimate was 210 pounds. The installed thrust estimate was provided by the propulsion subsystem evaluator, and was estimated as 57 pounds per engine, so that

$$\frac{T}{W} = \frac{2 * 57}{210} = 0.543$$

This served as the first estimate for thrust-to-weight for the MURV. This value was used in subsequent preliminary sizing studies and performance estimates. As the gross weight was refined for each configuration option, the thrust-to-weight changed, but was forced to change by the same amount for all configurations. In this way, though the final value for takeoff T/W differed from 0.543, it was constant among all candidate airframe designs.

2.1.3 Estimating Wing Loading at Takeoff, W/S With preliminary estimates of engine thrust and vehicle weight available, the wing was sized to be compatible with these parameters, and within the constraints identified for takeoff and landing on the 600 foot runway at the Jefferson Proving Ground in Indiana. The wing loading was constrained by the landing requirement, as shown in Figure 4.2 of Volume One. The desired wing loading was found by establishing a goal stall speed, V_{st}, and for a given value of $C_{L_{max}}$, W/S was calculated. The relationship of these parameters is given by

$$V_{st} = \sqrt{\frac{2(W/S)}{\rho C_{L_{max}}}} \qquad (2.5)$$

Rewriting, and solving for wing loading yields:

$$\frac{W}{S} = \frac{1}{2} V_{st}^2 \rho C_{L_{max}} \tag{2.6}$$

Assuming that the approach speed, V_{app}, is $V_{app} = 1.2 V_{st}$

$$\frac{W}{S} = \frac{V_{app}^2 \rho C_{L_{max}}}{(2)(1.44)} \tag{2.7}$$

A goal approach speed of 65 miles per hour was selected to give the operator a reaction time similar to that of driving an automobile. The maximum lift coefficient was estimated to be 1.6, based on a review of historical aerodynamic data for similar configurations [83:91].

With these values for $C_{L_{max}}$ and V_{app} the wing loading required was 12.0 pounds per square foot (psf). This value had to be compatible with the constraints depicted in Figure 4.2 of Volume One. There exists a range of solutions for μ_b and C_D which meets the landing distance constraint for the values of W/S and $C_{L_{max}}$ selected. Note that the C_L value in the constraint plot is that for approach, not $C_{L_{max}}$. A typical value for approach lift coefficient is about eighty percent of $C_{L_{max}}$ [83:10-15]. In this case, $C_{L_{app}}$ was 1.28. The possible combinations of μ_b and C_D which meet the landing constraint for these parameter values are shown in Figure 2.1.

Within the accuracy of these calculations, any of the combinations of μ_b, $C_{L_{app}}$, and C_D from Figure 2.1 were acceptable for the MURV design. For a wing loading estimate of 12.0 and the initial gross weight estimate of 210 pounds, the wing planform area needed to meet the landing constraint and allow an approach speed of about 65 miles per hour was 17.5 square feet.

2.1.4 Summary of Preliminary Sizing Parameters The preliminary sizing parameters are summarized in Table 2.2. These values were the initial sizing parameter values for the MURV conceptual external arrangements options. The development of these designs is described in the following section.

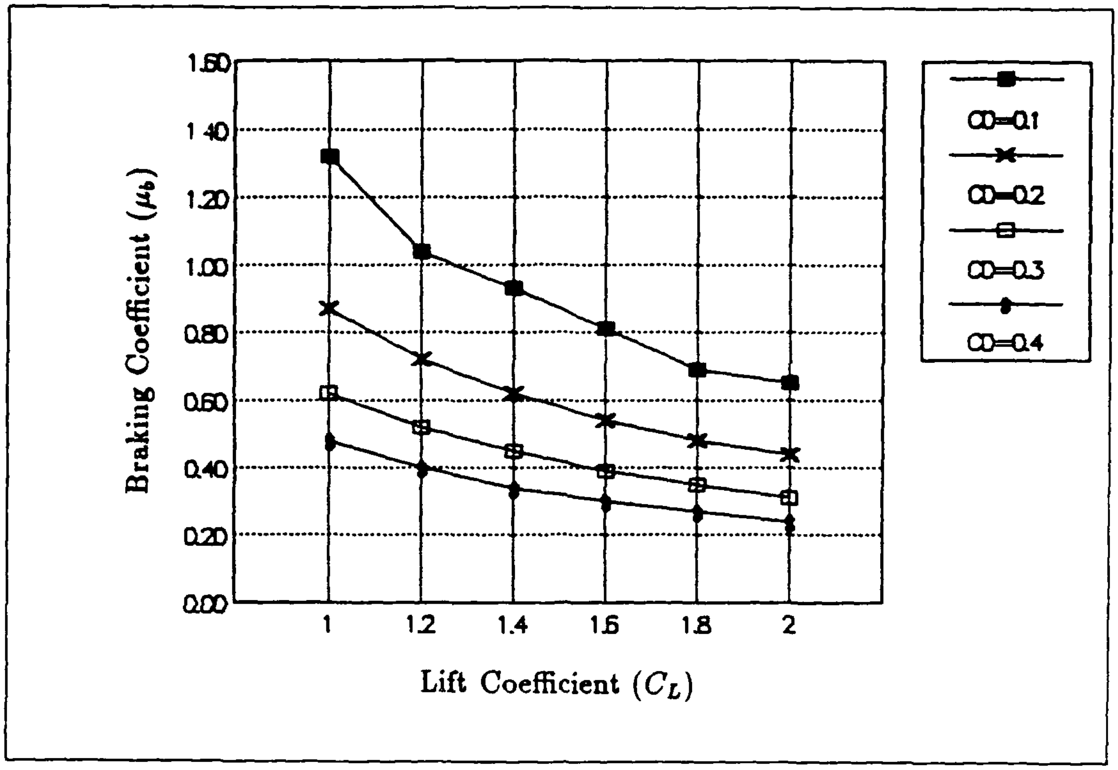


Figure 2.1. Solutions for μ_b and C_D which Meet Landing Constraints

2.2 *External Arrangement*

As was described in the Conceptual Design Phase, only concepts which had typical modern fighter characteristics were considered in the design of the external arrangement for the MURV. This meant that the wing planform and control surfaces had to have geometric shapes similar to those of typical modern fighters, and that the engine/airframe integration design features had to have the appearance of a fighter. The complete development of the wing planforms and geometries is presented here; however, the only propulsion system components discussed are those which affect

Table 2.2. Preliminary MURV Sizing Parameters

Parameter	Value
TOGW	210 lbs
T/W	0.543
W/S	12.0 psf
S	17.5 ft^2

the external shape and arrangement of the MURV, namely the air inlet and exhaust nozzle. All other external components are developed, such as the fuselage, vertical tail, and where applicable, horizontal tail or canard.

The approach taken to determine the external arrangement was to create a series of configuration options which were representative of typical modern fighter designs. The configuration items of interest were the fuselage shape, wing planform, wing sweep angle, and control surface arrangement and geometry. After reviewing dozens of fighter designs [33];[75];[84];[95], three basic configuration *layouts* appeared more prevalent in fighter designs than all others

Conventional: Trapezoidal wing planform with aft horizontal tail and vertical tail.

Wing/Canard: Delta wing planform with forward canard and vertical tail.

Tailless: Delta wing with vertical tail; no horizontal tail or canard.

A MURV conceptual design was created for each of these three fundamentally different configurations, and were given the following designations:

MURV-1: Delta wing; forward canard; vertical tail.

MURV-2: Trapezoidal wing; aft horizontal tail; vertical tail.

MURV-3: Delta wing; tailless; vertical tail.

The objective in evaluating three concepts was to select the one that was optimal for initial deployment of the MURV. Optimality was determined according to the objectives and measures of effectiveness discussed in Chapter IV of Volume One. In selecting the MURV airframe configuration, a subset of the complete list of measures of effectiveness was used containing those that were most critical to the aerodynamics, stability, and modularity of the MURV. These criteria are applied in Chapter XI, where the discussion of the selection process for the airframe is presented.

The configurations were developed with the supermaneuverability experiments as the primary design influence however, several limitations were placed on the candidate designs to isolate the benefits of each configuration and to simplify the iterative design process. The limitations were:

- No high-lift devices; i.e., no flaps, slats, etc. ...
- No ailerons, elevators, or trim tabs.
- No speed brakes.
- No ventral fins.

The design of these additional control surfaces was delayed until a single airframe was selected as the baseline configuration, at which time the various applicable control surfaces, wing, and tail/canard configuration were optimized.

2.2.1 Refined TOGW Estimate Since much of the information on the subsystem designs (flight computer, data aquisition system, sensors, etc.) was yet unknown, we assumed their weights and centers of mass were located equivalently for each of the three configurations. This assumed that all of the configurations would have similar electronic hardware components distributed throughout the vehicle in a similar pattern. Additionally, all three had identical engine, inlet, fuel system, and engine installation weights but at different station locations. The final consideration involved the "technology level" of the materials used for the load carrying structures. The technology level gave an indication of the percentage of weight of the vehicle structure devoted to composite materials. The most promising composite material for use in the MURV is KEVLAR. "Developed by the DuPont Company, KEVLAR has the highest strength-to-weight ratio of any commercially available fiber" [61:216]. Inherent to this high strength-to-weight ratio is a considerable savings in weight compared to fiberglass or aluminum structures. Because specific internal structure was not defined at the time of this analysis. an average technology level of 80% was assumed; i.e., we assumed a weight reduction of 20% below standard aluminum structure. We did not attempt to model actual wall thicknesses or strength of the structure, but used empirical methods to estimate the component weights and total vehicle weight. The methods are a function of many variables, but most significantly of the geometry of the major structural components (fuselage, wing, amd empennage).

2.3 Wing Planform Development

The review of modern fighter configurations led to establishing a range of wing aspect ratios and leading edge sweep angles consistent with most modern fighters. Aspect ratio, *AR*, is the ratio of the square of the wing span, b^2, divided by the wing planform area, S, and is a measure of the stubbiness of a wing. A high aspect ratio wing (*AR* greater than about five or six) will be long and thin like that of a Lockheed U-2, whereas a low aspect ratio wing (*AR* less than about four) appears short and stubby like a Lockheed F-104. From the range of aspect ratios found for most modern fighters, the MURV aspect ratio was restricted between 2.5 and 3.5 [84:142–148].

The leading edge sweep angle, Λ_{LE}, can theoretically be selected from the maximum design Mach number of the aircraft, M_{max}, by making use of the following relationship

$$\Lambda_{LE} > \cos^{-1}\left(\frac{1}{M_{max}}\right) \text{ (deg)} \qquad (2.8)$$

Using this relation, the Λ_{LE} calculated will be such that when the aircraft is at maximum Mach, the component of air velocity approaching the leading edge of the wing at normal incidence is just at sonic speed [60:280–283]. The purpose is to keep the wing leading edge in subsonic flow, and therefore lower drag, through as wide a Mach range as possible [75:Ch 7,13–15]. This criteria would indicate that an aircraft designed to Mach 2.0 should have a leading edge sweep of at least 60°. In practice, wing sweep angles are generally less than that calculated by Equation 2.8. In designing the MURV, we restricted the range of potential leading edge sweep angles from 45° to 65°. This would give the MURV an appearance of a fighter designed for a maximum Mach of 1.4 for the 45 degree sweep and 2.4 for 65 degrees. The sweep angle for each concept was chosen to be consistent with other modern fighters of similar wing planforms.

To calculate the aerodynamic performance of the wing (lift and drag), a nominal wing cross-section shape and maximum thickness had to be established. Both the wing cross-section and thickness effect the lift curve slope $C_{L_{\alpha_w}}$, $C_{L_{max}}$, and the angle-of-attack for $C_{L_{max}}$, $\alpha_{C_{L_{max}}}$ [75:Ch 7,1–13]. Since all three wings were designed with the same cross-section and to the same thickness, the values of the measures of effectiveness computed for each MURV were not skewed in favor of any particular design. The wing section, as it is normally referred, was selected from an extensive list of documented wing sections [77:484][75:App G,3–6]. The section aerodynamic characteristics depend on the wing thickness, t/c. A nominal thickness of 10% was chosen as a compromise between wing weight and aerodynamic performance. The section selected for the MURV was the NACA 64 series airfoil. This was chosen for its high lift curve slope and zero moment about its own aerodynamic center. These and other section characteristics are shown in Table 2.3.

Table 2.3. MURV Wing Section Characteristics (NACA Series 64 Airfoil)

$C_{L_{max}}$	1.17–1.2
$\alpha_{L=0}$	0.0
$C_{L_{\alpha_w}}$	0.104
$C_{M_{a.c.}}$	0.0

With the NACA 64 Series airfoil, the MURV has the inherent aerodynamic capability to generate high lift without the assistance of high lift devices such as flaps and/or slats. This does not negate the need for high lift devices; it does, however, alleviate the design requirements of such devices by relying on the wing to provide more lift at moderate angles-of-attack, which results in lower drag.

Table 2.4 summarizes the range of wing design parameters considered for the MURV.

2.3.1 Common Components The unique traits of each concept are its wing and control surface design and location. In order to identify the benefits of one over another, the impact of the

Table 2.4. MURV Wing Design Parameters

Parameter	Range
W/S	12.0 psf
S_w	17.5 ft^2
Λ_{LE}	45–65 deg
AR	2.5–3.5
Airfoil Type	NACA Series 64
t/c	0.1

unique elements had to be separated from all other influences. This was accomplished by designing the three configurations with a majority of common components, such as the inlet and fuselage. The only significant differences between the configurations were the wing design and location, and control surface design and position (aft tail versus canard).

In implementing this philosophy it became clear that not all other design parameters could remain the same for the three designs. For example, there were slight differences in c.g. location and inlet/engine placement; these were changed to normalize the stability parameter, C_{M_α}, as much as possible. But the major aerodynamic and stability design parameters, such as the wing loading, and thrust-to-weight remained approximately the same for all three designs. This is explained in further detail in the discussion of each MURV design.

The design philosophy and development of each concept is given in the introductory remarks to each design. The descriptions of the design discuss only those components which were unique to that configuration.

2.3.1.1 Inlet/Nozzle Design Once the restriction on engine type was determined, engine air inlets and exhaust nozzles had to be designed and integrated into the configuration, as well. For the MURV to be capable of performing supermaneuvers, the engine must operate at full capability throughout the ranges of angles-of-attack and sideslip expected. For maneuvers such as the Herbst maneuver, this could result in angles-of-attack greater than 70°. This places severe demands on the inlet design. Inlet efficiency degrades at higher angles-of-attack, and may be so poor as to cause engine surges or stalls [75:Ch 16,15–18]. One primary cause for the loss in efficiency is from aerodynamic disturbances created when the airflow entering the inlet cannot negotiate the severe change in flow direction, and *flow separation* is induced, which results in higher drag. Placing the inlet under the belly of the fuselage alleviates this problem somewhat by allowing the flow to be "straightened-out" by the fuselage before it enters the inlet. To take advantage of this phenomenon, all MURV concepts were designed with inlets located under the fuselage. The inlet size and shape was held constant as well, since these are dictated by the engine airflow demand and the arrangement of the internal fuselage volume, both of which were fixed elements of the three designs.

The exhaust nozzle was designed to allow gradual curvature from the main fuselage cross-section diameter to the exhaust nozzle exits. This was done to minimise the chance of flow separation on the nozzle contour, or boattail, hence reducing the drag.

2.3.1.2 Fuselage Design The fuselage was designed to have the appearance of a supersonic fighter, yet be modular in construction and easy to access and maintain. The parameters which determined the fuselage shape were the maximum diameter, d_{max}, and fineness ratio, l/d, where l is the fuselage length and d is the average diameter.

The fineness ratio was selected to be consistent with that of a modern supersonic fighter. The main factor in selecting the fineness ratio is the effect on drag [75:Ch 8,5–7]. The fuselages of supersonic fighter aircraft have fineness ratios from about eight to ten, so the MURV fuselage was designed to a fineness ratio of 9.0. This resulted in a fuselage length of about ten feet.

The maximum diameter was established by the size of the engines; that is, the maximum fuselage diameter occurred at the engine combustor cross-section. d_{max} was set for a side-by-side placement of two Microdynamics prototype engines, and allowance for clearance and cooling air between them and the sidewalls. This resulted in a maximum diameter of 13.5 inches. The corresponding maximum height was chosen to allow room for engine accessories, and for flexibility in wing and tail/canard mounting locations. This set the height to 12.3 inches.

2.3.1.3 Vertical Tail Section 4.5 of Volume One discussed the stability and control objectives applied during the Preliminary Design Phase. Recall that the longitudinal stability parameter, C_{M_α}, was used as the differentiating factor. A vertical tail with zero dihedral (oriented at 90° from horizontal) has a negligible effect on longitudinal stability. In fact, no term is included for the vertical tail in the expression for C_{M_α} [75:Ch 21,7]

$$C_{M_\alpha} = C_{L_{\alpha_w}} \frac{X_W}{\bar{c}} - C_{L_{\alpha_t}}(1 - \frac{\partial \epsilon}{\partial \alpha})\bar{V}_t + C_{M_{\alpha_i}} \tag{2.9}$$

where

$C_{L_{\alpha_w}}$	=	slope of C_L versus α for wing
X_w	=	distance from wing a.c. to aircraft c.g.
$\bar{c}$	=	wing mean aerodynamic chord
$C_{L_{\alpha_t}}$	=	slope of C_L versus α for tail
$\frac{\partial \epsilon}{\partial \alpha}$	=	change in tail downwash angle per change in α
$\bar{V}_t$	=	horizontal tail (canard) volume
$C_{M_{\alpha_i}}$	=	change in inlet moment coefficient per change in α

Since the vertical tail design (for zero dihedral) does not influence the longitudinal stability, its design did not have an impact on the measures of effectiveness for stability; therefore, all three MURV concepts were created with the same vertical tail volume, $\bar{V}_v$.

The tail volume value was selected to be representative of a typical fighter design. Vertical tail volume is given by

$$\bar{V}_v = \frac{S_v z_v}{S_w \bar{c}} \tag{2.10}$$

These terms are identical to those explained in Section 4.5 in Volume One, except here the moment arm distance, z_v, is the vertical distance from the vertical tail aerodynamic center to the aircraft c.g. Roskam [84:191-207] provides an historical data base of tail volume coefficients for several aircraft types, including fighters. From this data base we determined that an appropriate vertical tail volume for the MURV was 0.08. We redefined the vertical displacement to be twice the fuselage height to normalize the moment arm distance between all the concepts. Recall that the fuselage height was 12.3 inches, or 1.03 feet. The tail area required for a tail volume of 0.08 was found as:

$$S_v = 0.68\bar{c}$$

The area required to maintain a constant tail volume depended on the length of the wing mean aerodynamic chord. To alleviate this dependency, a nominal value for $\bar{c}$ was chosen to be three feet. This gave a tail area of 2.04 square feet. When the particular value for $\bar{c}$ was known for each concept, the actual tail volume was computed. Though it differed slightly between MURV-1, MURV-2, and MURV-3, it did not effect the decision making process.

All other performance measures concern the "point performance" of the MURV concepts, with the exception of range. Point performance calculations assume that the vehicle can be represented as a point mass, and include capabilities such as maximum speed, range, and endurance. The measures shown in Volume One, Table 4.9 represent point performance parameters, with the exception of $C_{M_{\alpha}}$. The only effect of the vertical tail was in its contribution to TOGW and drag. Since all tails had the same area, S_v, and were assumed to be made of the same material, they all contributed equally to the vehicle weight and drag estimates.

The shape of the vertical tail was common as well. The aspect ratio, AR_v, leading edge sweep angle, Λ_{LE_v}, and taper ratio, λ_v, were the same for all three concepts. The following values were set for these parameters based on the historical data base from Roskam:

$$AR_v = 1.04$$

$$\Lambda_{LE_v} = 45^\circ$$

$$\lambda_v = 0.32$$

Table 2.5 summarizes the vertical tail sizing parameters.

Table 2.5. MURV Vertical Tail Sizing Parameters

Parameter	Symbol	Value
Area	S_v	2.04 ft^2
Tail Volume	$\bar{V}_v$	0.08
Aspect Ratio	AR_v	1.04
Sweep Angle	Λ_{LE_v}	45°
Taper Ratio	λ_v	0.32

2.3.1.4 Canopy A canopy was added to give the appearance of a fighter and to provide additional internal volume, if needed. The canopy would be removable if it was not required for a particular experiment.

2.3.1.5 Summary of Common Components The inlet, nozzle, fuselage, and vertical tail were made common to all three designs to separate the effect of the wing/tail/canard concept from that of the subsystems. The design of these components is summarized in Table 2.6.

Table 2.6. Common Component Design Summary

Component	Design Factor	Implementation
Inlet	aerodynamic efficiency	located under fuselage
Nozzle	aerodynamic efficiency	shallow boattail angle
Fuselage	• supersonic design • engine integration	$l/d = 9$ $d_{max} = 13.5$
Vertical Tail	equivalent tail volumes	constant tail area (2.04 ft^2)
Canopy	similarity with full-scale fighter	—

In addition to these geometric similarities, the longitudinal stability was normalized as much as possible by placement of the inlet, engine, and internal fuel tank. The specific longitudinal stability parameters are presented within the sections relating to each design. C_{M_α} was the longitudinal stability parameter used as a measure of effectiveness, and is discussed in greater detail in Chapter XI of this volume.

2.4 The Three Concepts

After establishing the common components, the unique wing and tail/canard configurations were developed. For each alternative, the overall design philosophy is stated, followed by a description of the development of its unique elements. Table 2.8 at the end of this chapter summarizes the design and shape parameters for all three MURV design alternatives.

All three were created and analysed using a computer program contracted for the USAF Wright Aeronautical Laboratory's Technology Exploitaion Directorate, AFWAL/TADX, called the Interactive Design and Analysis System, or IDAS [58]. Appendix D gives a brief overview of the input required to examine a configuration using this program. The point to be made here is that, during the process of entering a configuration into the program, some geometric parameters were altered slighly due to the graphical interpretations made by IDAS. For this reason, the wing areas and some of the lengths were not exactly the same as intended, though the differences were negligible in the final analysis. For example, the wing areas for MURV-1, MURV-2, and MURV-3 were set to 17.07, 17.08, and 17.38 square feet, respectively, whereas the goal wing area was 17.5 square feet.

Slight differences arose in estimating the TOGW for the three concepts as well, due to the finer level of detail required by IDAS. The weight calculations were more detailed, requiring weight estimates for several subsystems still undefined. Although MURV-1 and MURV-2 were estimated at the goal gross weight of 210 pounds, the TOGW for MURV-3 was five pounds higher than the goal.

The combined effect of these differences was that the wing loading for the three concepts went from a goal value of 12.0 to 12.3, 12.3, and 12.36, respectively. These discrepancies did not significantly impact the analysis of the performace capabilities, however.

2.4.1 MURV-1 Design MURV-1 was designed with a delta wing and a forward canard. The Saab Viggen and the anticipated X-31 are examples of delta wing/canard designs. The main advantages of this type of configuration are in minimizing drag at cruise and high angle-of-attack by avoiding the additional drag penalty of elevator trim deflection, and to allow for reduced static stability at maneuvering conditions by using the canard as a control surface, not a stabilizing surface. This results from the fact that the wing aerodynamic center is aft of the c.g., thus the airplane is inherently stable when the canard does not produce lift. As the canard is deflected and produces lift, the aerodynamic center moves forward, and the aircraft becomes increasingly unstable, yet more maneuverable; thus, the canard is used to control the level of longitudinal stability of the aircraft.

An advantage of a canard over a conventional horizontal tail design is that the canard will not be affected by the wing's wake at high angles-of-attack as a tail can be. At such extreme conditions adequate control power is necessary to avoid loss of vehicle control and, if it does not exist, might result in the loss of the vehicle. Since the MURV is expected, and in fact specifically targeted, to perform such maneuvers, a canard design was very promising.

2.4.1.1 Wing Design Figure 2.2 shows the wing planform for MURV-1. The wing shape was determined by the leading edge sweep angle, Λ_{LE}, aspect ratio, AR, and taper ratio, λ. The sweep angle of 55° was selected as representative of a Mach 1.6-class delta wing fighter, and the aspect ratio of 2.74 gave a wing span of about 8 feet, which was chosen as a compromise of minimizing the wing span for transportability and maximizing it for aerodynamic efficiency. Note that Λ_{LE} and AR fall within the ranges identified in Table 2.4.

The ratio of the wing tip chord to the root chord lengths was 0.28. For this value of λ and the root chord length, c_r, the mean aerodynamic chord length, $\bar{c}$, was calculated as

$$\bar{c} = \frac{2}{3} c_r \left(\frac{1 + \lambda + \lambda^2}{1 + \lambda} \right) \tag{2.11}$$

For MURV-1, $\bar{c}$ was found to be 3.6 feet.

2.4.1.2 Canard The canard was mounted at the top of the fuselage for this particular drawing but, like the wing, could be mounted high or low. The high mounting here coincides with the low wing mounting to minimize aerodynamic disturbances between the canard and wing. The leading edge sweep angle was set at the minimum value of 45° to maximize the canard effectiveness, and the aspect ratio is approximately 2.68. The canard volume coefficient, $\bar{V}_c$, was set at 0.14. The optimal value is the subject of a trade study described in Chapter XI. The current value affected the decision criteria only through the canard weight and drag. These are relatively insensitive to small changes in canard planform area and moment arm.

2.4.1.3 Configuration Figure 2.2 shows the MURV-1 concept in three views. An airfoil section had been selected and the wing thickness established, as per Section 2.3. The engines were placed well aft to allow for integration of the inlet into and through the fuselage. An internal fuel tank sized to hold 65 pounds of fuel was inserted into the fuselage approximately at the c.g. to minimize the c.g. movement as fuel is burned.The figure shows a low wing mounting configuration, but the location can be either high or low due to the modularity features of the fuselage (see Chapter VII of this volume). From Equation 2.10 the vertical tail volume was found to be 0.07.

2.4.2 MURV-2 Design MURV-2 was designed to represent the more conventional wing/horizontal tail design, as seen on the F-15, F-16, and F-4 aircraft. This design uses the horizontal tail to trim out the aircraft pitching moment created by the wing, thus stabilizing the pitching motion.

A penalty associated with this type of configuration is that, at certain high-α attitudes, the air coming across the tail is often disturbed by the wing. At extreme conditions, the tail can lose all ability to stabilize the aircraft pitching motion. Therefore, an important design decision was whether to mount the tail in the same plane of the wing, above it, or below it. For MURV-2, the tail was located below the wing to minimize the effect of the wing wake to the tail aerodynamic power. Like MURV-1, the fuselage structure was designed for modularity so that the tail and/or

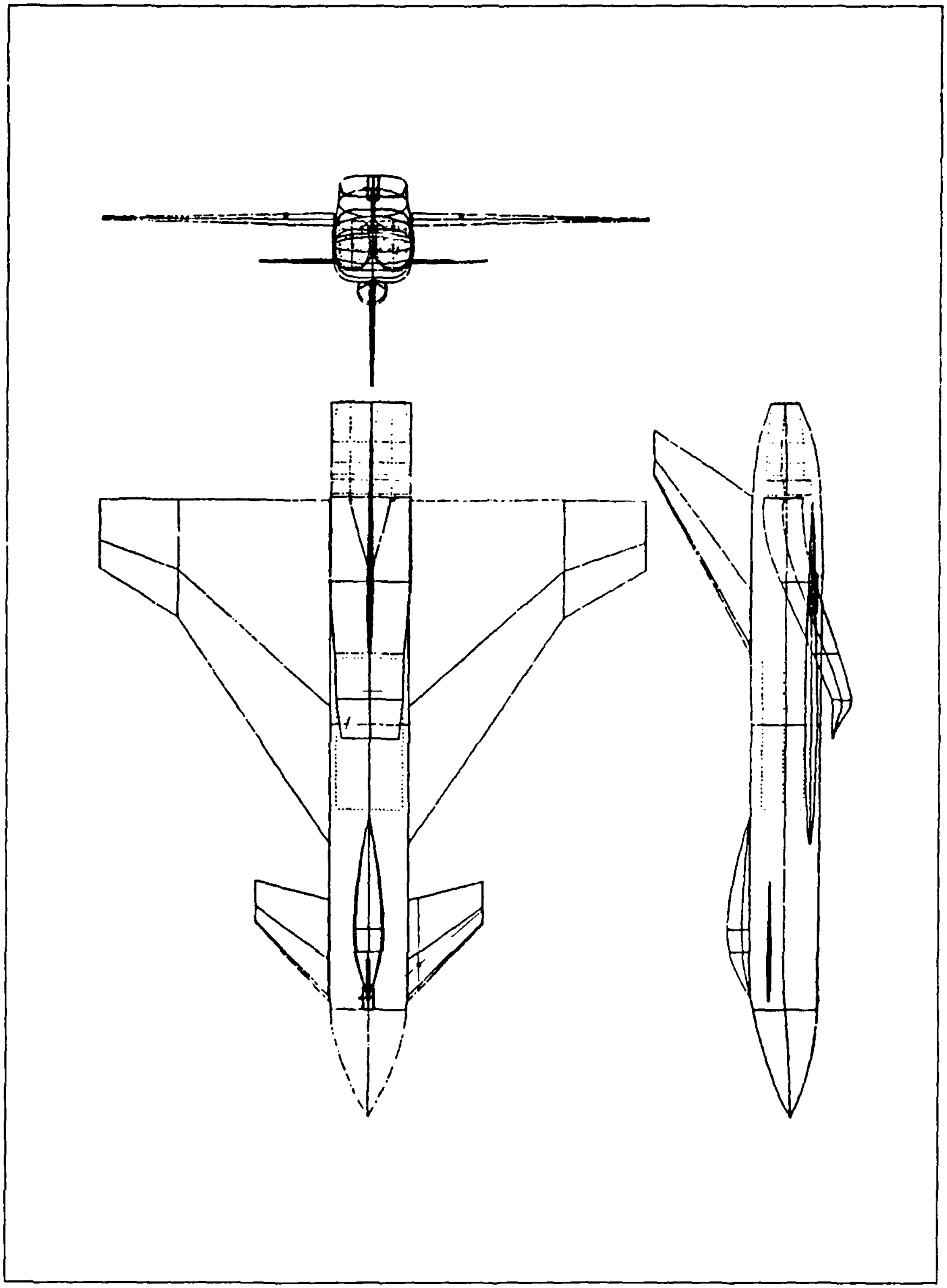


Figure 2.2. MURV-1 Configuration

the wing could be mounted high or low on the fuselage.

2.4.2.1 Wing Design The wing was sized according to Table 2.4. The wing planform is shown in Figure 2.3. As mentioned previously, the wing section and thickness were designed to be the same as for MURV-1, but the leading edge sweep angle was reduced to 45°. The aspect ratio of 3.2 was within the range specified in Table 2.4, and is representative of conventional fighter designs of this type. For the wing area of 17.08 square feet, the span was 8.1 feet. The root chord length was 3.6 feet, and the taper ratio, λ, was 0.32. The mean aerodynamic chord, $\bar{c}$, was found from Equation 2.11 to be 2.83 feet.

2.4.2.2 Horizontal Tail The horizontal tail was mounted at the bottom of the fuselage here but, as with MURV-1, could be mounted high or low. The low mounting was chosen to minimize the interference of the wing wake with the tail aerodynamics. The leading edge sweep angle was set at 45°, just as for the wing, and the aspect ratio is approximately 2.8. The tail volume was set originally at 0.3. The optimal value was not determined explicitly at this stage of the design, but was analyzed parametrically in Chapter XI. The tails were kept as far aft on the fuselage as possible to extend the moment arm and reduce the tail size needed for a given tail volume.

2.4.2.3 Configuration Figure 2.3 shows the conventional MURV-2 design in three views. For this design, the engine was moved slightly forward to keep the c.g. in an acceptable location, since the wing was moved well forward compared to MURV-1. The engine exhaust nozzle, or tailpipe, had to be extended about 14 inches to account for the more forward engine position. The internal fuel tank was moved forward so that it was centered approximately at the aircraft c.g. From Equation 2.10, the vertical tail volume which resulted from the mean aerodynamic chord length was 0.07.

2.4.3 MURV-3 Design MURV-3 was designed with a delta wing similar to that of MURV-1, but with no canard or horizontal tail. The Mirage 2000 and General Dynamics/Convair F - 106 and F - 102 are examples of delta wing, tailless designs. The main advantages of this type of configuration are in the drag and weight reduction realized with no other control surface. It also is an inherently stable design in the longitudinal mode, since the wing aerodynamic center is aft of the c.g., thus no stabilizing surface is required.

All control is performed by movable surfaces on the wing, either at the leading or trailing edge. The trailing edge must be capable of providing a negative camber shape to the wing to generate a positive pitching moment, otherwise leading edge devices are required.

2.4.3.1 Wing Design Figure 2.4 shows the wing planform for MURV-3. The wing shape is similar to that of MURV-1 except the leading edge is swept slighty more (60°) and the tip chord is 7.08 inches ($\lambda = 0.12$). The aspect ratio is similar to MURV-1 as well (2.64), resulting in a wing span of 7.83 feet. Again, note that all geometric parameters fall within the limits defined in Table 2.4.

For $\lambda = 0.32$ and a root chord of 5.07 feet, the mean aerodynamic chord length, $\bar{c}$, was calculated as 3.73 feet.

2.4.3.2 Configuration Figure 2.4 shows the MURV-3 concept in three views, with the airfoil section and wing thickness established as per Section 2.3. As in the design of MURV-2, the engines were moved further forward than for MURV-1 to locate the c.g. properly. The figure shows a low wing mounting configuration but, here also, the location can be either high or low. Obviously, there is no canard or horizontal tail volume to calculate for this design, but the vertical tail volume was calculated to be 0.06.

The exhaust nozzle extension of 14 inches was required here as in MURV-2, and the inlet duct shows a slight overlap with the internal fuel tank. Though the overlap could be avoided, it makes the point that the inlet duct design was more restrictive on MURV-3 than for the other two candidate designs.

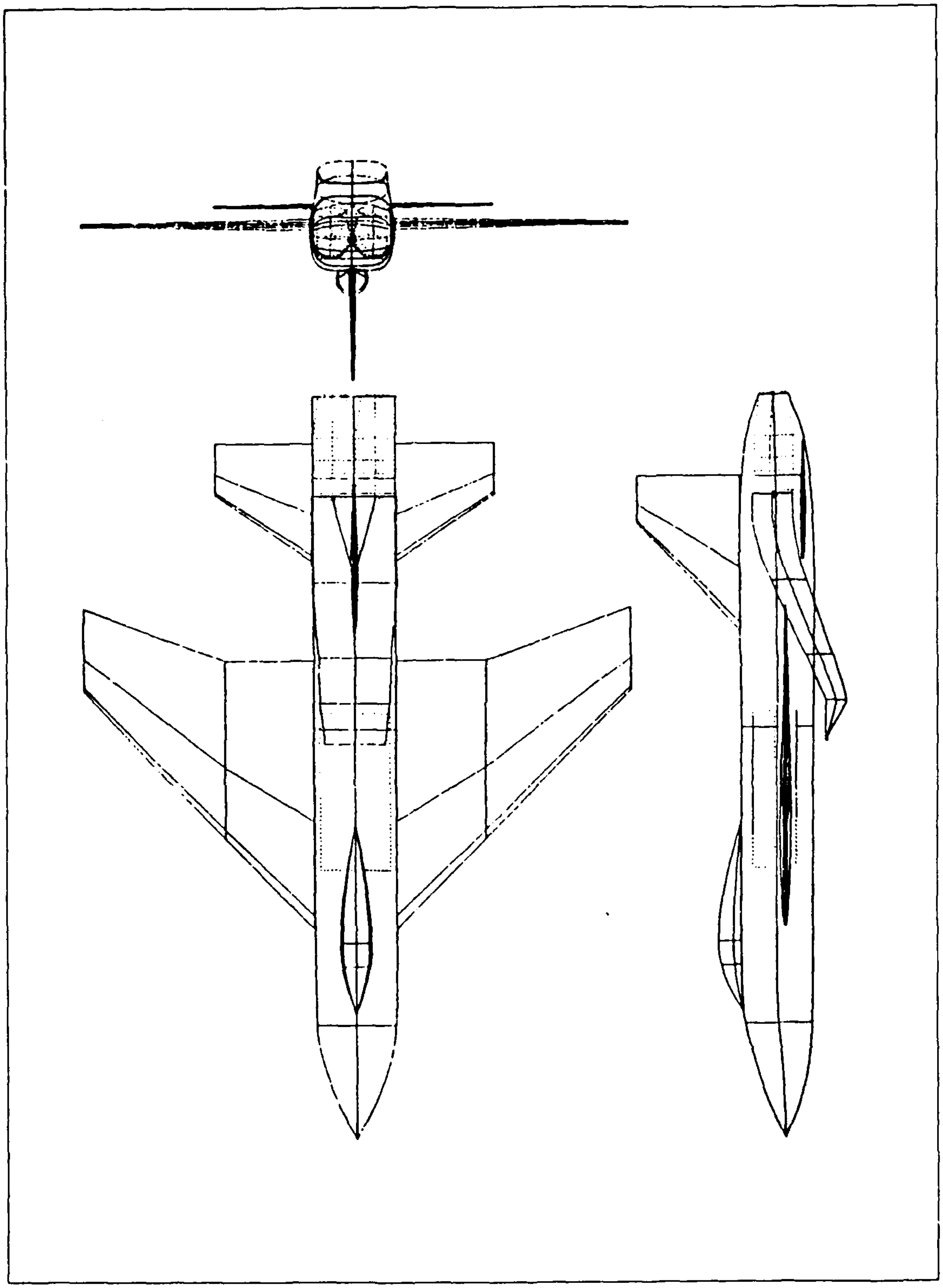


Figure 2.3. MURV-2 Configuration

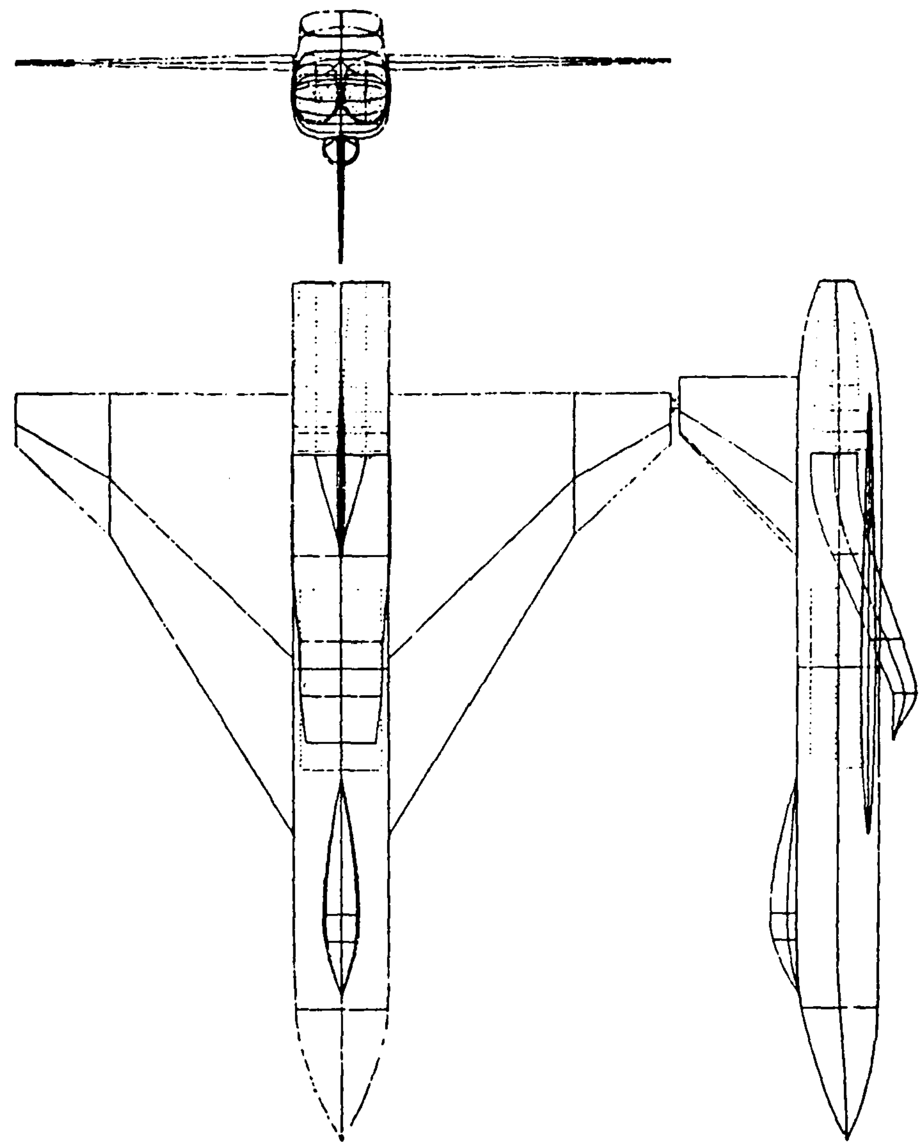


Figure 2.4. MURV-3 Configuration

2.5 Integrated Range and Mission Performance

The mission described in Section 4.4 of Volume One was executed with the goal of determining the measures described in that section. The inputs that produced the results are described in Appendix D. The results are listed in Table 2.7. The range and endurance are for a full throttle cruise at Mach 0.3 at 6000′ above sea level. The maximum Mach number, M_{max}, was determined by accelerating from Mach 0.2 to maximum velocity at 6000′, and the rate of climb and time to climb were based on climbing from 1000′ to 6000′ at full power.

These measures showed that MURV-1 was slightly better than the other designs, except for MURV-3's slight edge in range and endurance. Although the differences were too small to use these measures to differentiate between the options, the measures did show that all three alternatives have adequate performance in terms of range and velocity, since they all are able to reach a Mach number approaching the $M \approx 0.6$ recommended in Volume One, Section 4.4, and all are able to cruise for well over 30 minutes.

Table 2.7. Mission Performance Results

Measure	MURV-1	MURV-2	MURV-3	Units
Range	157	155	158	nautical miles
Endurance	48.5	48.0	48.8	min
M_{max}	.406	.402	.405	—
Rate of climb	78.0	76.6	76.4	ft/sec
Time to climb	1.51	1.48	1.50	min

2.6 Maneuver Performance

The *energy maneuverability* of the three concepts was computed to assess their maneuvering capabilities throughout their respective flight envelopes. According to Nicolai, energy maneuverability represents "...[the] aircraft's ability to change its energy state" [75:Ch 3,21]. The energy state of an aircraft is defined by the balance of its potential and kinetic energies. Potential energy (PE) is a function of weight and altitude while kinetic energy (KE) is a function of weight and velocity

$$E = PE + KE = Wh + \frac{WV^2}{2g} \tag{2.12}$$

Specific energy is the total energy per unit weight

$$e = h + \frac{V^2}{2g} \tag{2.13}$$

and is a constant for a given throttle setting and aircraft configuration [75:Ch 3,21–22]. Since maneuverability represents the dynamic change in energy, the time rate of change of e was needed. Differentiating Equation 2.13 with respect to time yields the specific excess power, or P_s

$$P_s = \frac{de}{dt} = \frac{dh}{dt} + \frac{V}{g}\frac{dV}{dt} \tag{2.14}$$

where dV/dt is the acceleration in the direction of the velocity vector. To make a comparison of the P_s values for the different concepts, it had to be expressed in terms of the fundamental forces acting on the vehicle which create the accelerations; that is, P_s had to be expressed in terms of lift, drag, thrust, and weight. Figure 2.5 shows a simplified free-body diagram of the longitudinal forces acting on an aircraft, where γ is the flight path (climb) angle, and i_T is the angle of the thrust vector with respect to the aircraft horizontal reference line (labeled WCL in the figure). For all MURV concepts, i_T was 0°.

Since the drag acts along the velocity vector, the acceleration in the direction of the velocity vector is found from Newton's Second Law as

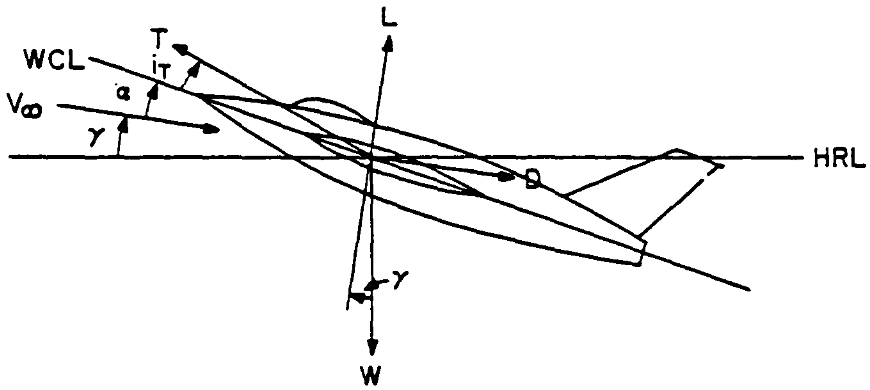
$$m\frac{dV}{dt} = T\cos\alpha - D - W\sin\gamma \tag{2.15}$$

Multiplying by V and dividing by W

$$\frac{V}{g}\frac{dV}{dt} = \frac{(T\cos\alpha - D)V}{W} - V\sin\gamma \tag{2.16}$$

Substituting Equation 2.16 into Equation 2.14, and noting that $\frac{dh}{dt} = V\sin\gamma$

$$P_s = \frac{(T\cos\alpha - D)V}{W} \tag{2.17}$$



[75:Ch 3, 2]

Figure 2.5. Free-Body Diagram of Aircraft Forces

which relates the time rate of change of energy, P_s, to the forces acting on the aircraft. P_s varies throughout the flight envelope as T, D, V, and W change. When $P_s = 0$, $T = D$, and the aircraft has no excess thrust to climb or accelerate. As α increases, the lift and drag increase as well, so that to maintain a constant speed and altitude, the thrust must equal or exceed the higher drag. Therefore, P_s represents a measure of the ability of the aircraft to maneuver at high angles-of-attack, accelerate at a constant altitude, and climb at a constant speed, where a higher value means greater capability.

We computed the P_s values of the three concepts throughout their respective flight envelopes. Maneuver conditions were examined by determining P_s for increased angles-of-attack by recalculating the weight while pulling n gs, often referred to as a *load factor* of —em n, as $W_g = nW$. Figure 2.6 shows contours of P_s versus Mach number and altitude for the three concepts at load factors of 1,3 and 5. The outermost contour of each chart is for $P_s = 0$, and represents the sustained speed and altitude limits at that load factor. The basis of comparison was the relative extent of the flight envelopes and the magnitude and pattern of the P_s contours. Note that there is no significant difference in any of the contour levels and patterns at comparable flight conditions. This was expected since all three candidates were designed to the same T/W and wing loading levels.

2.7 Summary

This section has described the development of the size and shape for three fundamentally different MURV concepts. The size was quantified by the gross weight (TOGW), wing loading (W/S), and the thrust-to-weight ratio (T/W). These parameters were constrained by the requirement to takeoff and land at the 600 foot runway at Jefferson Proving Ground, Indiana. All three concepts were sized to the same wing loading and the same engine thrust.

The shapes were representative of three major classes of fighter configurations: (1) delta wing with canard (MURV-1); (2) trapezoidal wing with tail (MURV-2); (3) delta wing with no tail/canard (MURV-3). Though the wing and control surface layouts were different, all three were designed with common fuselage, inlet, and nozzle components. The Integrated Design and Analysis System program was used to help create and analyze the designs. Table 2.8 summarizes the primary design parameters for the three MURV concepts.

The dynamical similarity parameter, Π_{DS}, was computed for each of the three designs. The values are shown in Table 2.8 below. Note that, though MURV-2 has the greatest dynamic similarity with with respect to a full-scale F-16, all three designs are similar to within one order of magnitude ($\Pi_{DS} < 10$). This suggests that, so long as the MURV dimensions are not drastically reduced, it and all configuration variations of it will produce dynamic data which can be applied, with proper judgment and caution, to full-scale aircraft.

The airframe concept represents a single component of the MURV system, though perhaps the

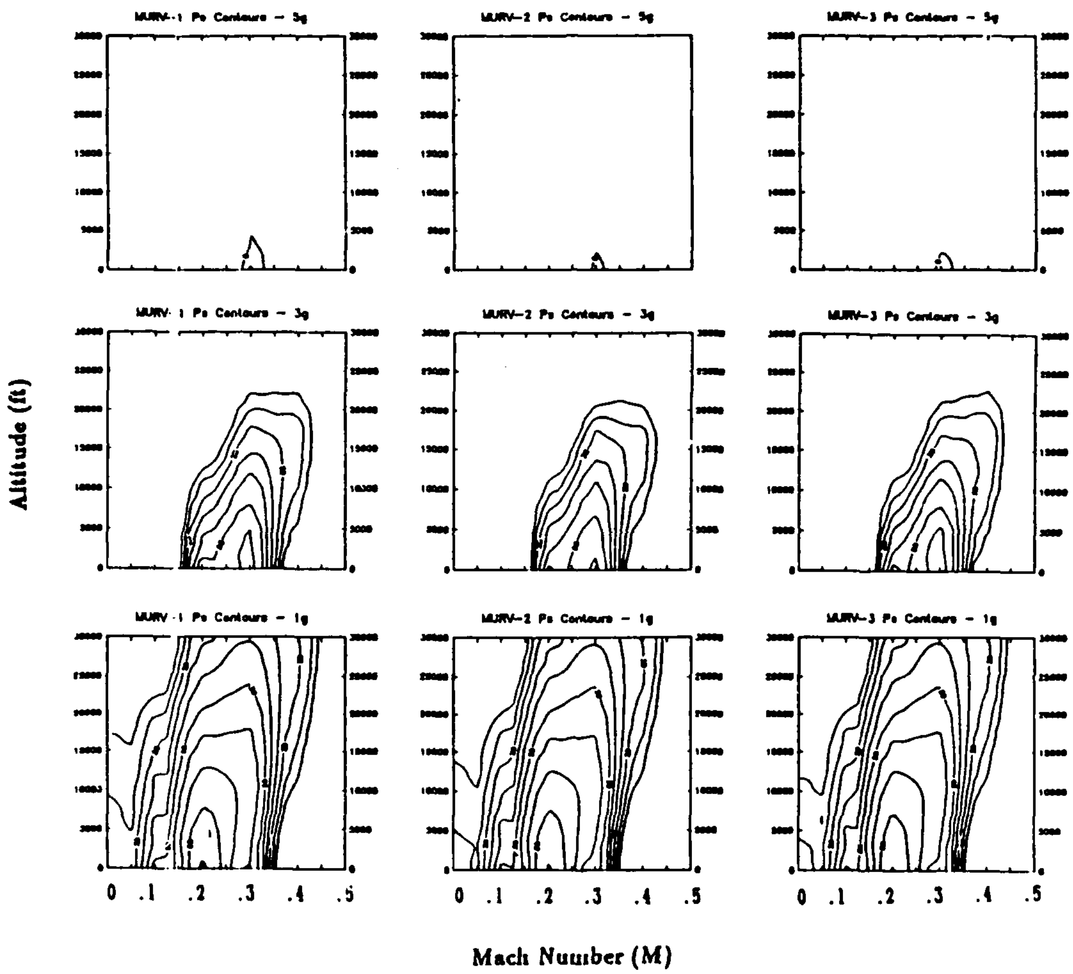


Figure 2.6. P_s Contour Maps for Candidate MURV Designs

Table 2.8. MURV Concept Design Summary

Parameter	MURV-1	MURV-2	MURV-3
TOGW (lb_f)	210	210	215
Length (ft)	10.00	10.00	10.00
Span (ft)	7.80	8.10	7.83
Height (ft)	2.97	2.97	2.97
T/W	0.543	0.543	0.53
W/S (psf)	12.30	12.30	12.36
S_w (ft^2)	17.07	17.08	17.38
Λ_{LE} (deg)	55	45	60
AR	2.74	3.20	2.64
$\bar{c}$ (ft)	3.60	2.83	3.73
$\bar{V}_{t/c}$	0.143	0.340	—
$\bar{V}_v$	0.07	0.09	0.06
Π_{DS}	6.825	5.976	8.163

most visible. The sections which follow present the development of the remainder of the subsystems, such as the engine, flight control system, and data acquisition system. Criteria relating to many of these areas were applied in selecting a single airframe to carry into the next phase of the external configuration development, described in Volume One, Section 5.1. Therefore, the description of the decision making process for the airframe selection is given in Chapter XI at the end of this volume. The aerodynamic and stability analyses of these configurations are presented as part of that process.

III. Propulsion System

A remotely piloted vehicle is a complex system, made of many components. The engine is a major system which must must be carefully integrated into the rest of the system to maximise the system performance. The main items considered in the preliminary design stage of propulsion integration were the selection of candidate engines, and the design and analysis of inlets with an emphasis on modularity. Designs of the fuel supply system, electrical power system, and the engine starting system were also discussed. The first task was to narrow the choice of engines to be considered for use in the MURV design. This was accomplished in three iterations. The first iteration of the selection process determined the type of powerplant to use for the vehicle. The types of powerplants considered ranged from ducted fan reciprocating engines to turbojet engines. The next screening was used to reduce the number of alternatives within the engine type selected in the first iteration. These engines were used in subsystem development analysis and evaluated in terms of aircraft performance. The final engine recommendation was determined on the basis of the results from the aircraft configuration studies and the overall needs of the MURV.

3.1 Preliminary Engine Selection

It is critical to have an engine of the correct size to match the mission of the remotely piloted vehicle. The proposed missions for the MURV include diverse tests ranging from high angle of attack post stall maneuvers to inlet/exhaust engine nozzle modifications to severe battle damaged flight control compensation. The engine must provide sufficient thrust to carry out the range of proposed tests as well as allowing for growth for future unforeseen thesis projects with an emphasis on modularity. The sizing and scaling of the aircraft depended on many factors, some of which were thrust, weight and physical size of the engine. The size of the aircraft was roughly limited due to cost and vehicle gross weight which was discussed in Section 2.1.2. A literature search was done on existing aircraft powerplants with a thrust between 40 and 300 pounds. This sise range was picked during an initial screening of the problem to design an aircraft of a size which could perform the intended mission but not be too large and costly. The determination of the upper sizing limit was described in Section 2.1.2.

3.1.1 Engine Type Determination The design study focused on a modular air vehicle configuration which can perform supermaneuverability type experiments. This led to an aircraft with characteristics similar to a modern air-to-air fighter. To accomplish these types of experiments, the aircraft needed to have a thrust to weight ratio of 0.75–1:1, which is a high ratio. The modular unmanned research vehicle must have the flexibility to accomplish many different experiments having extreme flight condition requirements. This translates into an overpowered aircraft with a large thrust to weight ratio. The vehicle also has a heavy payload requirement which has led to a larger aircraft and helped to size the vehicle and scale the upper limit of static thrust needed for the design. another factor which went into the requirements of the propulsion system was that there was some interest in inlet and nozzle testing as missions for the aircraft. During the conceptual design, these considerations narrowed the powerplant requirements from the wide variety of engine types available to those which produce jet-like thrust.

3.1.1.1 Propulsion System Needs, Objectives, Measures of Effectiveness, and Constraints The needs, objectives, and measures of effectiveness for the first iteration of screening were broad in nature to help identify a class of engines to pursue into further analysis. The engine was required to:

1. have a compact size to allow as much flexibility in fuselage sizing as possible,
2. have a low fuel consumption to minimize the weight and volume of fuel needed,
3. have a high thrust to weight ratio to increase the aircraft's overall thrust to weight ratio,
4. be fully throttleable, and
5. have a high reliability.

These needs were directly paralleled by subsystem objectives and measures of effectiveness. The objectives (with measure of effectiveness labels in parentheses) are listed below:

1. To minimize size, measured by maximum diameter and length (size).

2. To minimize fuel consumption, measured by the specific fuel consumption (SFC) at sea level static conditions.

3. To maximize thrust to weight ratio, measured by sea level static thrust divided by engine weight (F_g/W_e).

4. To maximize throttleability, measured by whether the engine could be operated at partial power at sea level static conditions (part power).

5. To maximize reliability, measured by the complexity of the engine (complex). A simple engine is less likely to fail during flight.

There were a few constraints on the engine which were also applied during the research for engines and during this iteration of screening possible classes of engines. As stated above the engine had to produce jet-like thrust to be considered. The engine had to be built with existing technology, but this did not exclude engines in the last stages of development. The engine had to be large enough to power the aircraft. The lower limit was chosen to be 50 pounds static thrust (lb st) because of the required payload weight as was discussed in Section 2.1.2. Multiple engine designs using the smaller engines were considered during the analysis. The last constraint was that engine data was readily available from the manufacture. These constraints, objectives and measures of effectiveness were used to narrow the selection of available engines which are discussed in the next section.

3.1.1.2 Thrust Producing Engines Several types of thrust producing engines were evaluated for use in the MURV: turbojets, turbofans, ducted fans, and pulsejets. The first type of engine considered was the turbojet. There was a large range of turbojet engines available but only a limited number in the low thrust category. The available engines ranged from 40 lb st to the 300 lb st upper limit imposed by the scope of this investigation [99:186–7]. The domestic engines available are listed in Table E.1 in Appendix E. Foreign manufactured engines were researched but dropped from consideration during the design process. They were eliminated because of the unavailability of manufacture's performance information in a reasonable time. A list of foreign produced engines available is listed in Table E.2 in Appendix E. The Microdynamics and Teledyne engines were in final stages of development, but they were considered since the final component selection for the MURV will be made in the follow-on detailed design phase of the project.

The next type of engine looked at was the turbofan engine. Turbofan engines are the most prevalent type used in the aircraft industry because of their high power to weight ratio and superior fuel consumption rate compared to turbojets, but no turbofan engine was found with a thrust less than 300 pounds.

Ducted fan engines were also considered for the design. This type of powerplant is a shrouded reciprocating engine driving a multibladed fan. A ducted fan looks like a jet engine and is rated in pounds of thrust. These powerplants are most commonly found in small model jet aircraft. Ducted fans currently in production are too small to meet the needs of the MURV.

Another type of powerplant considered for the aircraft was the valveless pulsejet. This type of engine is simple in design with no moving parts and ideal for small applications. The valveless pulsejet consists of a diffuser at the intake, geometry which forces more gas to be ejected from the rear than the front during combustion, a combustion chamber, and a tuned length tailpipe. The disadvantages of this type of engine are the noise and vibration produced, the length needed for the tuned length tailpipe, and the low efficiency of the engine. The pulsejet would not be suitable for engine nozzle experiments because of the uneven air flow through the engine. The typical specific fuel consumption (SFC) of a pulsejet engine is approximately 5 lb fuel/hr/lb thrust compared to less than 1.2 lb fuel/hr/lb thrust for a typical small turbojet engine [95:973]. The pulsejets currently produced are listed in Appendix E.

The types of engines listed above were the only ones found to meet the requirements of the MURV. The evaluation of the these categories of powerplants is done in the next section.

3.1.1.3 Selection of Engine Type Because of the reasons given in the previous section, the only feasible categories of engines for the MURV were turbojets and pulsejets. These two types were compared against each other using the measures of effectiveness listed above. Each engine was given an overall rating for each measure with a + being superior, 0 being neutral and a – being inferior. For this selection all objectives were weighted equally, and a + was assigned as +1 and a – was assigned a –1. The numeric values were then summed to find the overall merit for the category. The results are given below in Table 3.1. The results from the table above show that the turbojet engine was the best choice to pursue for use in the vehicle. A list of suitable turbojet engines follows in Table 3.2.

Table 3.1. MOE Ratings for Engine Categories

Engine	Size	SFC	F_g/W_e	Part Power	Complex	Total
Turbojet	+	+	+	+	–	+3
Pulsejet	–	–	–	–	+	–3

Table 3.2. MURV Candidate Low Thrust Turbojet Engines

Manufacturer	Designation	Thrust (lb st)	Weight (lb)	Diameter (in)	Length (in)
Williams Int'l	WR24-7	190	44	11.9	19.5
	WR24-8	240	50	11.9	19.5
Teledyne CAE	Model 312	135	30	8.3	14.0
	Model 320	200	40	9.9	16.8
Microdynamics	N/A	70	10	6.0	6.5

[99:186-7]

3.1.2 Turbojet Engine Selection The selection of the turbojet engine to be carried on into further analysis was based upon the refined needs of the MURV. In addition to the needs described earlier in Section 3.1.1.1, the powerplant needed to:

1. have a simple and safe fuel system,
2. provide electrical power generation capability to power the on-board electronics, sensors and actuators,
3. be low in cost, and
4. have a low development risk.

These needs listed above and those in Section 3.1.1.1 led to the propulsion subsystem objectives and measures of effectiveness used to evaluate the alternative engines listed in Table 3.2. The objectives and measures of effectiveness (with the measure labels in parentheses) are listed below:

1. To minimize size, measured by the length (L_e) and installation cross section (CS_e) which was found by multiplying the maximum width times height of the engine.
2. To minimize fuel consumption, measured by the specific fuel consumption (SFC) at sea level static conditions.
3. To maximize thrust, measured by the gross sea level static thrust (F_g).
4. To minimize weight, measured by engine weight (W_e).
5. To maximize simplicity of the fuel system, measured by the type of system needed (fuel). The alternatives had an external pump or an internal pump with a low pressure feed.
6. To maximize power generation, measured by kilovolt-amps (KVA).
7. To minimize developmental risk, evaluated in terms of the production history of the engine. An engine which was currently in production was given a higher rating than an engine in development (risk). The alternatives were assigned a rating of low, medium or high.

The measures of effectiveness were used to determine how well the engines met the objectives during the evaluation process.

The small turbojet engines were all similar in performance specifications. The comparison and selection was based upon off the shelf performance and configuration. The engine selected for the vehicle will have to be tailored in areas such as instrumentation, installation and starting systems. These modifications were independent of the engine selection because they can be performed on any of the alternatives considered and were not used as criteria for the decision. The acquisition costs for the engines were not used for the selection process because of the variability in the the cost estimates. The estimates were directly related to the size of production runs of the engines. How widely the engines will be produced was unknown during the time of this study. The Microdynamics engine was preproduction priced at an estimate of $15,000 while the Teledyne engine prices ranged from $15,000 for an adapted engine from a full production run to $40,000 for a one of a kind preproduction prototype [2]. The selection process used will be described next.

3.1.2.1 Analytical Hierarchy Process The selection process used for this iteration of engine selection was the analytical hierarchy process using normalized geometrical means to calculate the rating vectors [59]:

$$\hat{u}_i = c \prod_j (a_{ij})^{\frac{1}{n}} \tag{3.1}$$

where

$\hat{u}$ = the rating vector

c = a normalizing constant

a_{ij} = an element from the judgment matrix $\mathbf{A}$

n = dimension of the judgment matrix

The a_{ij} element in a judgment matrix reflects the comparison of the ith row item and the jth column item. For example, if the row item is mildly better than the column item, the element a_{ij}

would be a 3. The rankings are relative and a 3 does not mean the row item is three times better than the column item. If the column item is slightly better than the row item, then the element of the matrix would be the reciprocal, 1/3. A more detailed explanation of the process is contained in Appendix G

3.1.2.2 Measure of Effectiveness Ranking The most important measures were SFC and KVA. Specific fuel consumption is a measure of how efficient the engine is and was highly weighted because of its effect on the aircraft's thrust to weight ratio. The higher the SFC, the lower the number of pounds of fuel required to fly a mission. KVA was deemed important because of the large amount of electrical power needed to operate the flight control and data acquisition systems. Power generation from the engine is more efficient in terms of KVA per pound than alternative power sources such as batteries. The next measure in importance was the amount of thrust produced by the engine because of the hight thrust to weight ratio objective. A more powerful engine is more flexible in terms of accomplishing different types of experiments. Engine weight, W_e, was important to help minimize the overall vehicle weight. The next measures of effectiveness were used to measure the compactness of the engine size, CS_e and L_e. The installation cross section of the engine was used to make a fairer comparison between the engines. The installaton clearance was found by multiplying the maximum width times the height of the engine. This was a better measure than the maximum engine diameter because the installation cross section compensates for large engine protrusions. The installation cross section was given a higher weighting than the engine length because it was a main driver of fuselage cross sectional area. The engine length was used to penalize an engine which was smaller in cross section but longer than the rest of the alternatives, because a shorter engine will require less fuselage structure and weight. The smaller the size of engine the more flexibility there is in designing and modifying the fuselage, so a more compact engine is better. The last two measures of effectiveness used were FUEL, and RISK. The type of fuel system was important because a system which needed a fuel pump was more complex and had a higher weight than a system which did not require a pump. Production risk was used to categorise the inherent danger of relying on a system which was still in development such as the Microdynamics or Teledyne engines. The judgment matrix comparing the importance of the measures is in Table 3.3. The measure comparison matrix was then used to compute the criterion scale by the method of normalized geometric means using Equation 3.1. This vector is shown in Table 3.4.

Table 3.3. Engine Measure Comparison Matrix

MOE	SFC	KVA	F_g	W_e	CS_e	L_e	FUEL	RISK
SFC	1	2	3	4	4	4	6	7
KVA	1/2	1	2	4	4	4	6	7
F_g	1/3	1/2	1	2	2	2	5	6
W_e	1/4	1/4	1/2	1	2	2	4	5
CS_e	1/4	1/4	1/2	1/2	1	2	4	5
L_e	1/4	1/4	1/2	1/2	1/2	1	4	5
FUEL	1/6	1/6	1/5	1/4	1/4	1/4	1	2
RISK	1/7	1/7	1/6	1/5	1/5	1/5	1/2	1

Table 3.4. Engine Selection Criterion Scale

MOE	Weighting
SFC	0.3022
KVA	0.2415
F_g	0.1427
W_e	0.1009
CS_e	0.0849
L_e	0.0751
FUEL	0.0305
RISK	0.0222

3.1.2.3 Engine Ranking and Selection Each alternative was compared against the others for each measurement of effectiveness. The comparison was done with the alternative judgment matrices. As an example, the judgment matrix for engine installation cross section is shown below in Table 3.6. The Williams WR24-7 has a diameter of 11.9 inches, but an oil sump causes the height of the engine to be 15.5 inches. The Microdynamics engine was the smallest with a diameter of 6.0 inches, but a width of 13.0 inches was used because of the twin engine installation. To make a fair comparison of the engines, the installation cross section was used to compensate for the different clearances required by the engines. The values for installation cross section which were compared are shown in Table 3.5. The WR24-7 was first compared with the other alternatives. The WR24-8 has the same cross section so the relative rating was given a 1. When compared against the Teledyne Model 312's cross section, the rating of 1/9 was assigned to reflect that the smallest cross section was strongly preferred. For the opposite comparison, when the 312 was compared against the WR24-7, the reciprocal rating of 9 was assigned. The process continued until every engine was compared against the others. Judgment matrices for the other measures of effectiveness are in Appendix F. The next step was to calculate the rating vectors from the alternative judgment matrices using Equation 3.1. The rating vector for installation cross section is in Table 3.7.

Table 3.5. Engine Installation Cross Sections

Alternative	Width (in)	Height (in)	Cross Section (in^2)
WR24-7	11.9	15.5	184.5
WR24-8	11.9	15.5	184.5
312	8.3	8.3	68.9
320	9.9	9.9	98.0
Micro	13.0	6.0	78.0

Table 3.6. Engine Alternative Comparison Matrix for Installation Cross Section

Alternative	WR24-7	WR24-8	312	320	Micro
WR24-7	1	1	1/9	1/7	1/8
WR24-8	1	1	1/9	1/7	1/8
312	9	9	1	3	2
320	7	7	1/3	1	1/2
Micro	8	8	1/2	2	1

Table 3.7. Installation Cross Section Rating Vector

Alternative	Rating
WR24-7	0.0367
WR24-8	0.0367
312	0.4395
320	0.1941
Micro	0.2930

The rating matrix was compiled by combining the measure rating vectors as columns. This formed the 5×8 matrix shown in Table F.10 in Appendix F. The rating matrix was multiplied by the criterion scale to get the overall rating for the five engine alternatives. The results of the calculation of the overall rating is in Table 3.8. As can be seen from this table, the two engines which best fit the propulsion needs of the MURV are the Teledyne Models 312 and 320. These engines will be carried forward into the next iteration of design and used for subsequent subsystem development. The inlet design and integration is covered next.

Table 3.8. Overall Ratings for Engine Selection

Alternative	Overall Rating
WR24-7	0.1722
WR24-8	0.1376
312	0.2714
320	0.2246
Micro	0.1946

3.2 Engine Integration

Integration of the engine into the aircraft design involves many areas. The areas which will be covered in this section are the inlet design methodology, preliminary inlet design and optimization, installed thrust and drag computations, fuel system design, electrical power considerations, and the starting system requirements. The specific subsystem designs are also discussed in Section 5.2. Engine location and mounting concerns are discussed in Section 7.2.2

3.2.1 Inlet Design Methodology An aircraft's performance is based upon many factors. The engine has an important role in this performance and the inlet design is a key factor. The preliminary inlet design was based upon the inlet configuration, efficiency estimate, and the inlet drag. The baseline missions for the MURV are a family of supermaneuverability experiments, which includes post stall maneuvering at a high angle of attack (AOA) flight condition. To meet the engine airflow needs at very high angles of attack, the inlet must be located in the freestream air. This post stall maneuvering also involves high turn rates at low airspeeds which can cause loss of air induction on a side mounted inlet. This drove the requirement of an inlet located on the bottom of the fuselage. No other inlet locations were considered feasible because of the supermaneuverability missions. However, other problems can occur with a bottom mounted air inlet. An improperly located bottom inlet could be shielded from the airflow at a negative angle of attack causing a compressor stall or flaming out the engine. Either of these problems could cause the flight test to be aborted or even the loss of the aircraft. Ingestion of debris is a concern during ground operations with any jet aircraft, but the problem is compounded with a bottom mounted inlet. The launch/recovery system must be designed to help prevent debris from being kicked up into the inlet and sized to keep the inlet far enough away from the ground. The special provisions to shield the inlet are described in Section 5.5.2.3.

The main purpose of the inlet is to take air from the freestream conditions and route it to the engine at the right conditions with minimum pressure losses. For the preliminary design, the inlet was designed using one dimensional compressible isentropic flow. The naming convention for duct stations in this section will be a 0 or ∞ for freestream conditions, hl at the inlet highlight, 1 at the duct throat and 2 at the duct exit or engine face as shown in Figure 3.1. Because the subsonic inlet can draw air from a larger freestream area (A_0) than the inlet highlight area (A_{hl}), variable geometry is not needed and was not considered.

3.2.1.1 Inlet Throat and Capture Areas The first step in designing a subsonic inlet was to determine the throat area needed to supply the correct amount of air required by the engine. The throat Mach number was the first variable to be picked. The lower the throat Mach number, the larger the inlet and the higher the efficiency. In subsonic inlet design, the throat Mach number corresponding to choked flow is approximately 0.9, and the throat area should be sized so the Mach number does not exceed 0.8 [68:356]. A lower throat Mach number value of 0.7 was used for this design to help increase the pressure recovery efficiency, since the aircraft will only fly at low Mach

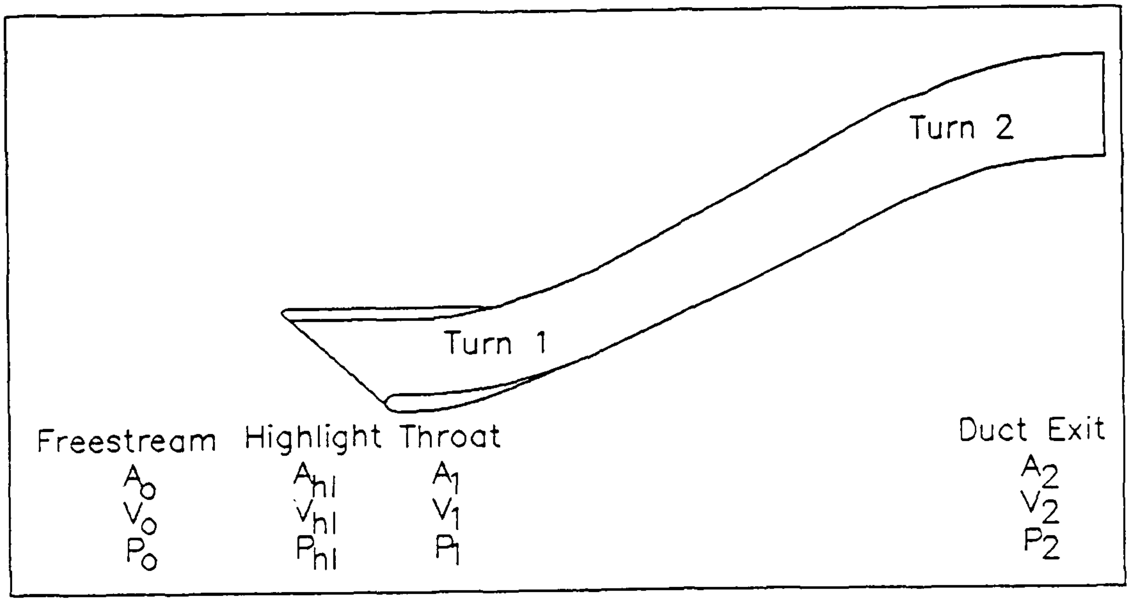


Figure 3.1. Inlet Duct Station Naming Convention

numbers. This value also allows for some engine growth and is far enough away from the critical value to prevent choked flow. The throat area (A_1) was found at the chosen throat mach number using the equation

$$A_1 = \frac{\dot{m}\sqrt{T_{t_0}}}{P_{t_0}\,\mathrm{MFP}} \tag{3.2}$$

where

$$
\begin{aligned}
\dot{m} &= \text{engine mass flow} \\
T_{t_0} &= \text{total temperature} \\
P_{t_0} &= \text{total pressure} \\
\mathrm{MFP}(M,\gamma,R) &= \frac{M\sqrt{\frac{\gamma g_c}{R}}}{\left(1+\frac{\gamma-1}{2}M^2\right)^{\frac{\gamma+1}{2(\gamma-1)}}} = \text{mass flow parameter} \\
M &= \text{Mach number} \\
\gamma &= \text{ratio of specific heats} \\
g_c &= \text{gravitational constant} \\
R &= \text{gas constant of air}
\end{aligned}
$$

The mass flow in the inlet is a combination of the air required by the engine and the bypass air needed to cool the engine. For the purposes of the conceptual inlet design, the amount of cooling mass flow used for throat sizing was determined by the amount of air required to change out the volume of the engine compartment every second. A detailed analysis will need to be accomplished in the detailed design phase after the final engine selection is made and temperature data is available from the engine manufacturer. The mass of bypass air was determined by subtracting the volume of the engine from the estimated volume of engine compartment and multiplying the result by the air density. A thermal blanket is also proposed to protect the fuselage skin from the engine heat. This is discussed in Section 7.2.2.

The next step was to determine the highlight area, A_{hl}, which is the area bounded by the points most forward on the inlet lip. This is also known as the capture area. For typical conventional subsonic inlets, the ratio of the highlight diameter to the throat diameter, $\frac{D_{hl}}{D_1}$, ranges from 1.10 to 1.16 [68:358]. The larger the ratio the less flow separation will occur on the inside wall of the inlet. The major effect of flow separation and distortion is low efficiency and a possible compressor stall. To accomplish the supermaneuverability experiments, the vehicle will operate at high angles of attack and with rapid yaw rates. These conditions are much more severe than those a conventional aircraft face, so the highlight diameter ratio was increased to 1.20 to allow for less distortion around the lip while operating at extreme flight conditions. The increased capture area size also helps the inlet pressure recovery when the aircraft is operated with the freestream area larger than the capture area, $\frac{A_0}{A_{hl}} > 1.0$; such as when the aircraft is operating on the ground or in a sustained climb. After the inlet areas were found, the lip shaping must be addressed.

3.2.1.2 Inlet Lip shaping The shape of the inlet lip is important because it determines how the air will flow around the leading edge of the inlet. A sharp lip will induce separation at much lower angles of attack than will a blunt lip. A bell mouth inlet which is typically used during ground testing of engines during development has little separation but would not be practical because of the drag caused by the large capture area. The MURV will be a slow flying aircraft and a blunt type of leading edge was picked for the lip shape. The leading edge of the inlet was formed with a quarter circle cross section to connect the throat to the highlight. The bottom of the inlet leading edge was connected to the highlight by a quarter circle and the top was connected with an ellipse. An ellipse was used to reduce the top lip thickness reduced by one fourth. The top of the inlet did not need to be as thick as the bottom and sides due to the interaction of the airstream with the fuselage. At high angles of attack, the airflow at the top of the inlet will be reflected off of the bottom of the fuselage into the inlet, and the airstream will not be as likely to separate upon entering the inlet as the airstream entering the bottom or sides. To help with the high angle of attack flight conditions a 45 degree sweep extension was added to the top of the inlet lip to increase the capture area. The degree of sweep reflects a Mach two class of aircraft inlet, keeping with fighter-like appearance of the design. This extension will also act as a boundary layer splitter. A drawing of the inlet lip shape is in Figure 3.2.

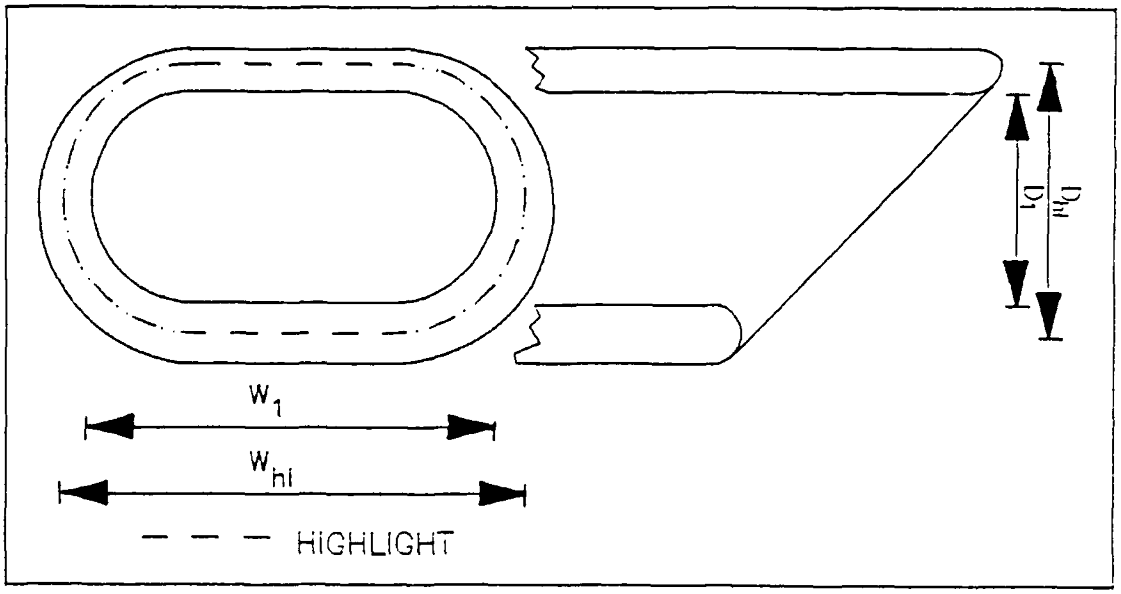


Figure 3.2. Inlet Lip Shaping

3.2.1.3 Highlight Area Shaping The geometric shaping of the throat and highlight areas were driven by a few needs. The inlet should be short in height to reduce the necessary aircraft ground clearance and the length of the landing gear. At least two inlet diameters of clearance from the ground is recommended by Nicolai [75:Ch 16,18]. Inlet efficiency is a function of the highlight perimeter and the internal wetted surface area of the duct, so the circumference should be minimised. The inlet also had to fit between the mounting rails on the fuselage, so width was a constraint. Sharp corners also can induce pressure losses and affect the efficiency. A circular shape is the most efficient shape with the smallest perimeter for a given surface area, but the size of the diameter increased the vertical offset and the length of the landing gear. An oval shape was generated using two semicircles connected with a rectangular section. This shape reduced the required height but retained smooth lines and a reasonable perimeter to area ratio. A width to height ratio of 2.0 was used for the oval shape to reduce the overall height of the inlet. A thin inlet has the advantage of having a low profile and reducing the drag behind the inlet but it can cause distortion or choking in the flow from separation inside the inlet. A wider inlet has the advantage of having a larger capture area at high angles of attack than a thinner inlet.

3.2.1.4 Boundary Layer Diverter Standoff For peak performance, the engine needs to have air with the least distortion possible. The low energy turbulent air from the fuselage boundary layer should be avoided from being drawn into the inlet. A boundary layer diverter was designed to add a standoff to the inlet from the fuselage to prevent the ingestion of this low energy air. This standoff also helps to keep the inlet from being shielded from the airstream during negative AOA conditions. The first step in designing the boundary layer diverter was to determine the flight conditions where the aircraft would have the largest boundary layer at the inlet highlight. An airspeed of Mach 0.1 at sea level was chosen as a representative point for the boundary layer diverter calculations because this is about as slow as the aircraft will be able to fly. The Reynolds number was calculated at these freestream conditions using the equation

$$Re_x = \frac{\rho_\infty V_\infty x}{\mu_\infty}$$

where

ρ_∞	=	air density
V_∞	=	air velocity
μ_∞	=	absolute viscosity coefficient
x	=	length of flat plate friction

The boundary layer thickness (δ_t) was calculated using turbulent flow because this was a larger value and more conservative than for laminar flow. The following equation approximates the turbulent boundary layer thickness [1:125]:

$$\delta_t \approx \frac{0.37x}{Re_x^{0.2}}$$

The boundary layer diverter standoff from the fuselage was taken to be $2.0 \times \delta_t$ as was recommended by Nicolai [75:Ch 17,15].

3.2.1.5 Inlet Geometry The vertical offset (Δy) of an inlet is the distance from the center of the engine face to the center of the inlet opening. The length (L) of the inlet is the distance horizontally from the engine face to the highlight. D is the diameter of the inlet duct at the engine face. The Δy required for the inlet was determined by summing the following lengths:

maximum engine radius

installation clearance

height of fuselage cover

height of boundary layer diverter

half height of inlet

---

vertical offset

As a starting point for the duct length, the following ratios for a conventional diffuser were used [92:12]:

$$\frac{\Delta y}{D} = 1.5 \tag{3.3}$$

$$\frac{L}{D} = 3.5 \tag{3.4}$$

This led to the relationship $L = 2.33\Delta y$. The duct length affects the efficiency of the inlet. The lower the efficiency of an inlet the less installed thrust an engine can produce. A short inlet can cause separation and a long inlet has a higher friction loss. A shorter inlet was also desired to reduce possible interference with fuel tank placement during a configuration change. These considerations were addressed during inlet optimization and fuel tank design.

The routing of the air from the duct throat to the engine face will complete the discussion of the inlet geometry. The air was routed from the throat along a one inch straight section before the first turn was initiated. The two duct turns were of equal arc length and radius to simplify design and analysis. The air flow was routed up into the fuselage through a straight diffuser which connected the two duct turns. The final section of the duct connected turn two to the air bypass section near the engine face with a one inch piece to straighten the flow before it enters the engine. Bypass cooling air is bled here with an two inch inner sleeve directing the engine demand air toward the compressor face. The bypass air is routed around the inner sleeve and outside the engine to cool the engine and is exhausted out around the tailpipe. Uniform diffusion was used with the area constantly increasing from the throat to the exit of turn two. The height of the inlet was also uniformly increased from the throat to the bypass air interface. This is shown in Figure 3.3.

3.2.1.6 Inlet Efficiency A measure of efficiency for an inlet is η, the pressure recovery:

$$\eta = \frac{P_{t_2}}{P_{t_0}}$$

Optimization of the duct was concerned primarily with increasing η by varying the length of the duct to determine the best length of inlet for the aircraft. The inlet optimization is described in

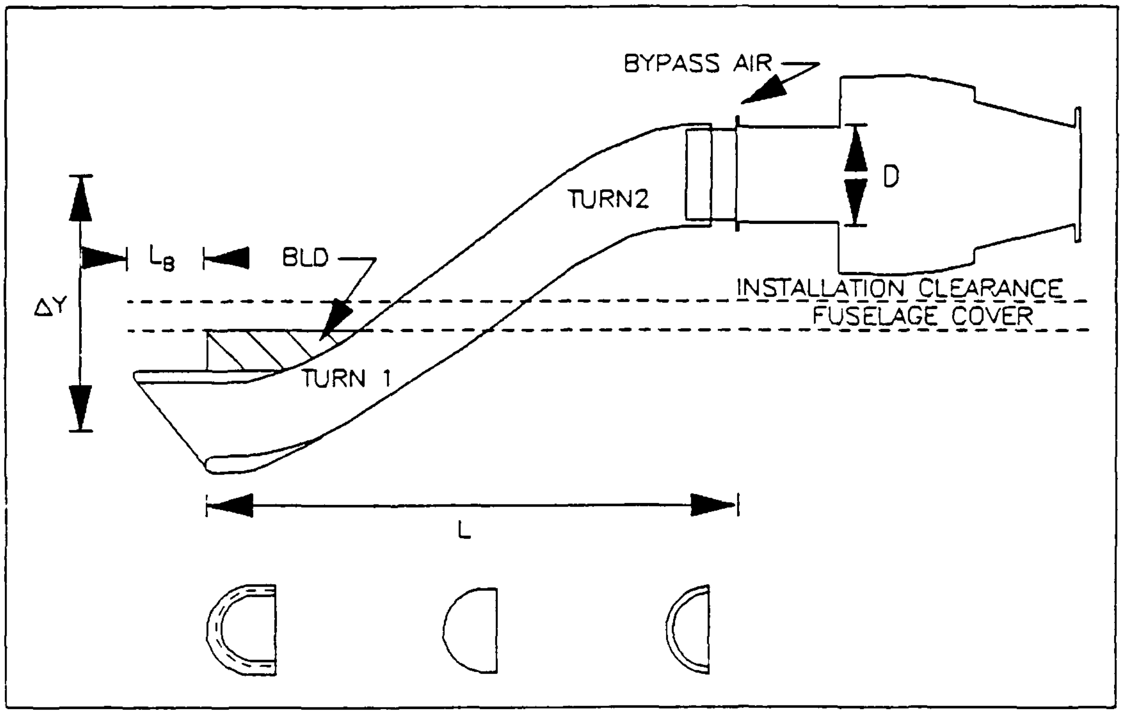


Figure 3.3. Inlet Geometry

Section 3.2.3. The inlets were evaluated in terms of the pressure recovery, which was calculated using the computer program SIRPP from Aeronautical Systems Division/ENFTA [52]. The inlet geometry is input, and mass flows and pressure recoveries at various Mach numbers are calculated. The program accounts for the major losses that occur in a subsonic inlet. These losses are inlet lip losses, duct offset losses, duct turning losses, and diffusion losses. The geometric input parameters are listed in Appendix H and shown graphically in Figure H.1. Sample output for SIRPP is also listed in Appendix H. The values for η were used to determine the installed thrust as well as for inlet analysis purposes.

3.2.1.7 Inlet Drag Estimation The components of inlet drag which were analyzed are the additive and boundary layer diverter drags. The additive drag is the loss of momentum of the stream tube of air defined by the capture area, A_{hl}, that is diverted around the inlet. Some of the loss may be recovered by lip suction. The boundary layer diverter drag is caused by the momentum lost by the air as it is turned by the BLD. When $\frac{A_0}{A_{hl}} < 1.0$, the additive drag coefficient (C_{D_A}) can be found using the following equations [75:Ch 17,4]:

$$
\begin{aligned}
C_{D_A} &= C_{D_{Add}} K_{Add} \\
C_{D_{Add}} &= \frac{P_0}{q_\infty}\left(\frac{A_1}{A_{hl}}\right)\left[\left(\frac{P_1}{P_0}\right)(1+\gamma M_1{}^2) - 1\right] - 2\left(\frac{A_0}{A_{hl}}\right) \\
\frac{P_1}{P_0} &= \left(\frac{P_1}{P_{t_1}}\right)\left(\frac{P_{t1}}{P_{t_0}}\right)\left(\frac{P_{t0}}{P_0}\right) \\
\frac{A_0}{A_1} &= \frac{\left(\frac{A}{A^*}\right)_{M_0}}{\left(\frac{A}{A^*}\right)_{M_1}} \\
\frac{A}{A^*} &= \frac{1}{M}\left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M^2\right)\right]^{\frac{\gamma+1}{2(\gamma-1)}}
\end{aligned}
$$

where

$C_{D_{Add}}$ = uncorrected additive drag coefficient

K_{Add} = an empirical correction term for lip suction

q_∞ = dynamic pressure

A^* = the area corresponding to Mach flow

The boundary layer diverter drag coefficient ($C_{D_{BLD}}$) was found using the following equations:

$$C_{D_{BLD}} = \left(\frac{C_{D_{BLD}}}{2\theta_{BLD}}\right)(2.6\theta_{BLD})\left(\frac{A_{BLD}}{A_{hl}}\right)$$

where

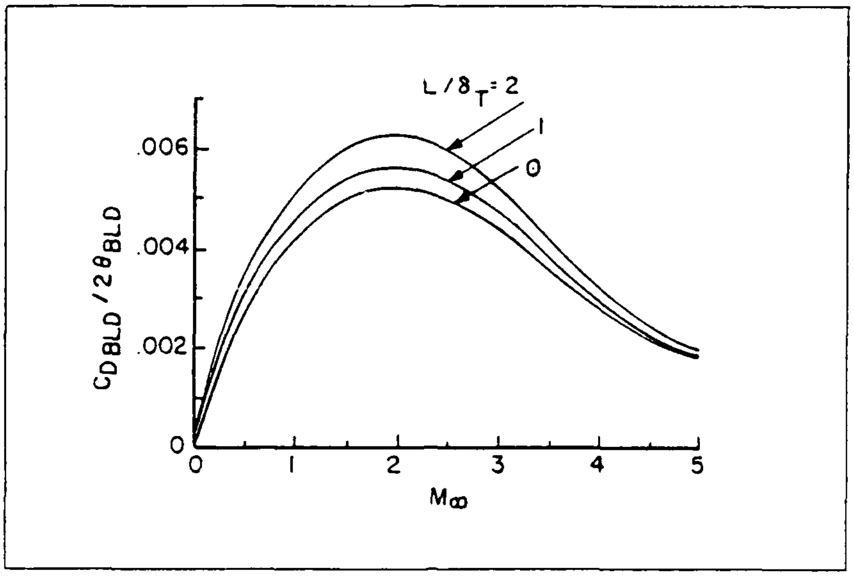
$\frac{C_{D_{BLD}}}{2\theta_{BLD}}$ = a value obtained from Figure 3.4

L_B = diverter offset length defined in Figure 3.3

θ_{BLD} = diverter compression ramp angle

A_{BLD} = projected surface area in flow direction

3.2.2 Preliminary MURV Inlet Designs The inlet design process described above was used to determine the preliminary inlets for the Teledyne engines. The Model 312 is described first



[75:Ch 17,13]

Figure 3.4. Boundary Layer Diverter Drag Variation with Mach Number

because of the higher rating from second iteration of the engine selection process. The throat sizing was the first concern addressed during the design process. The bypass air was found from the difference of volumes between the engine compartment and the engine. For the preliminary design, the engine compartment was approximated with a square cross sectional area with sides of 10.25 inches and a length of 11 inches. The square cross section of the engine compartment allowed for a one inch installation clearance. The boattail tapered from a width of 10.25 inches to 5.25 inches around the tailpipe with a length of 14 inches. These dimensions gave v-'ume of 2026 in^3. The approximate engine volume of 635 in^3 was subtracted to leave a volume of 1391 in^3. This was multiplied by the sea level air density to get the amount of air required to change out the volume of the engine compartment once per second. The bypass air used in the throat sizing was 0.061 $\frac{lb_m}{s}$. The throat area was found for the maximum $\dot{m}$ which occurred at the highest altitude at the lowest airspeed, 30,000 feet at mach 0.2. The area of the throat from Equation 3.2 was 9.15 inches. The highlight diameter was found using a scale factor of 1.2 to give 13.18 inches. The boundary layer diverter standoff distance was found using two times the turbulent boundary layer height at the flight conditions of mach 0.1 at sea level altitude with the inlet being located at approximately five feet from the nose. The distance for the standoff using these conditions was two inches. The vertical offset, Δy, was found by summing the following:

$$\begin{aligned}
\text{maximum engine radius} &= 4.125 \\
\text{installation clearance} &= 1.000 \\
\text{height of fuselage cover} &= 1.500 \\
\text{boundary layer diverter} &= 2.000 \\
\text{half height of inlet} &= \underline{1.475} \\
\Delta y &= 10.10
\end{aligned}$$

The vertical offset, Δy, was multiplied by 2.33 to obtain the duct length, $L = 23.53$ inches. The radius of the turns was $\frac{1}{2}L = 11.8$. The inlet drag was estimated next.

The drag for the inlet was calculated using the equations in Section 3.2.1 at freestream conditions of mach 0.4 at sea level. Because of the high throat Mach number, the additive drag term was very small. The lip suction correction coefficient for area ratio, $\frac{A_0}{A_{hl}}$ was approximately 0.001 [75:Ch 17,7]. This led to negligible additive drag. The boundary layer diverter drag coefficient was found using a θ_{BLD} of 20.29 degrees, an A_{BLD} of 9.76 in^2 and a L_D of 3.07 inches which was equal to the height of the inlet,

The inlet efficiency was measured by η, which was calculated using the SIRPP computer program. The sea level static inlet pressure recovery for full throttle was 0.917. The same process was used to design a preliminary inlet for the Model 320 engine. The higher mass flows led to a larger inlet with increased additive and boundary layer drags. The major inlet parameters for the Teledyne engines are summarized in Table 3.9. The preliminary inlets will be optimized in the next section.

Table 3.9. Preliminary MURV Inlet Parameters

Parameter	Model 312	Model 320
A_{hl}	13.18 in^2	19.22 in^2
W_{hl}	5.43 in	6.56 in
D_{hl}	2.72 in	3.28 in
A_1	9.15 in^2	13.34 in^2
W_1	4.53 in	5.46 in
D_1	2.26 in	2.74 in
A_2	13.20 in^2	19.63 in^2
BLD	2.00 in	2.00 in
Δy	10.10 in	11.23 in
L	23.53 in	26.19 in
Turn radius	11.80 in	13.10 in
C_{D_A}	Negligible	0.0004
DRAG$_A$	Negligible	1.82 lb$_f$
A_{BLD}	9.76 in^2	10.60 in^2
θ_{BLD}	20.29°	18.32°
L_B	3.07 in	3.69 in
$C_{D_{BLD}}$	0.1289	0.0999
DRAG$_{BLD}$	2.80 lb$_f$	3.16 lb$_f$
η	0.917	0.909

3.2.3 Inlet Length Optimization The preliminary inlets for the Teledyne engines were optimized to increase the efficiency, and the pressure recoveries were determined using the SIRPP program for inlet lengths ranging from 23 to 50 inches. The highlight and throat shaping and dimension. were held constant as was the radius of curvature for the turns. Less turning was required with a longer inlet. the increased length reduced the turning losses but increased the friction losses. The losses were measured by the differential pressures for diffusion, turning and friction:

$$\frac{\mathrm{DP}}{q} = \frac{P_{t_2} - P_{t_1}}{q_1}$$

The differential pressures obtained from the SIRPP runs for the various inlet lengths are shown graphically in Figures 3.5 and 3.6 and summarized in Tables I.1 and I.2 found in Appendix I. The sealevel static pressure recoveries for the inlets are shown in Figure 3.7 As shown in the figures, the total pressure losses are minimized with a length of 35.5 inches for the Model 312 and 38.2 inches for the Model 320. The Models 312 and 320 had thrust increases of 1.38 percent and 1.27 percent using the optimized inlets.

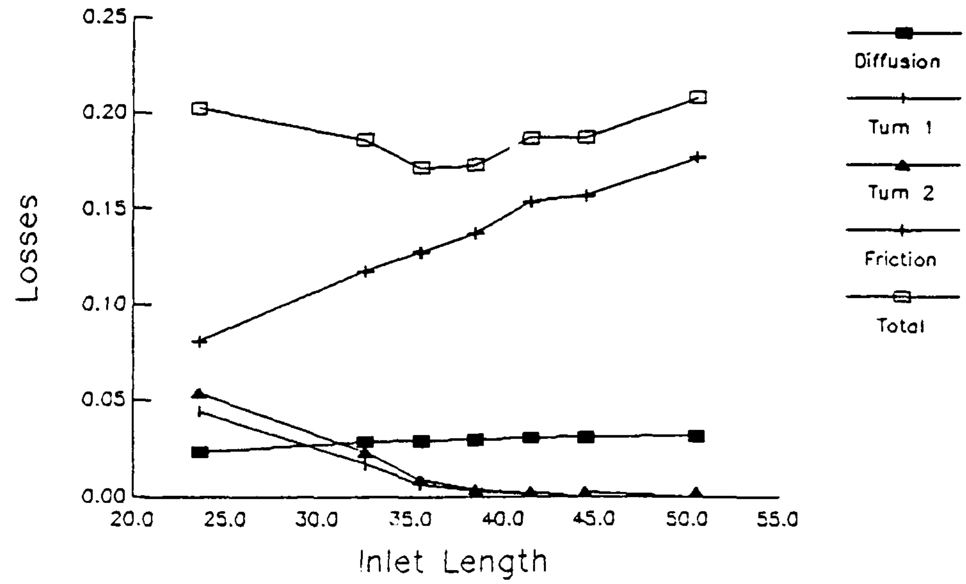


Figure 3.5. Model 312 Inlet Differential Pressures

The inlets will slightly infringe on the fuel tank as located in the preliminary aircraft design discussed in Section 2.2. To minimize the impact, the fuel tank could be built with a cutout for the inlet or the tank and engine could be shifted slightly during the detailed design to prevent conflict. The inlet could also be shortened to the point where it does not impact the fuel tank location, but because of the need for the maximum installed thrust from the engine, the inlet was kept at the optimum length. The fuel tank preliminary design is discussed in Section 3.5.1. The next area of integration which will be discussed will be nozzle considerations.

3.2.4 Nozzle Considerations The design mission of the MURV is to perform supermaneuverability types of experiments. Thrust vectoring is one way to produce side forces which are necessary for some advanced supermaneuverability experiments such as the Herbst maneuver. A thrust vectoring nozzle is not part of the aircraft's baseline design, although provisions have been made for later efforts in this area through modularity. The boattail can easily be redesigned to

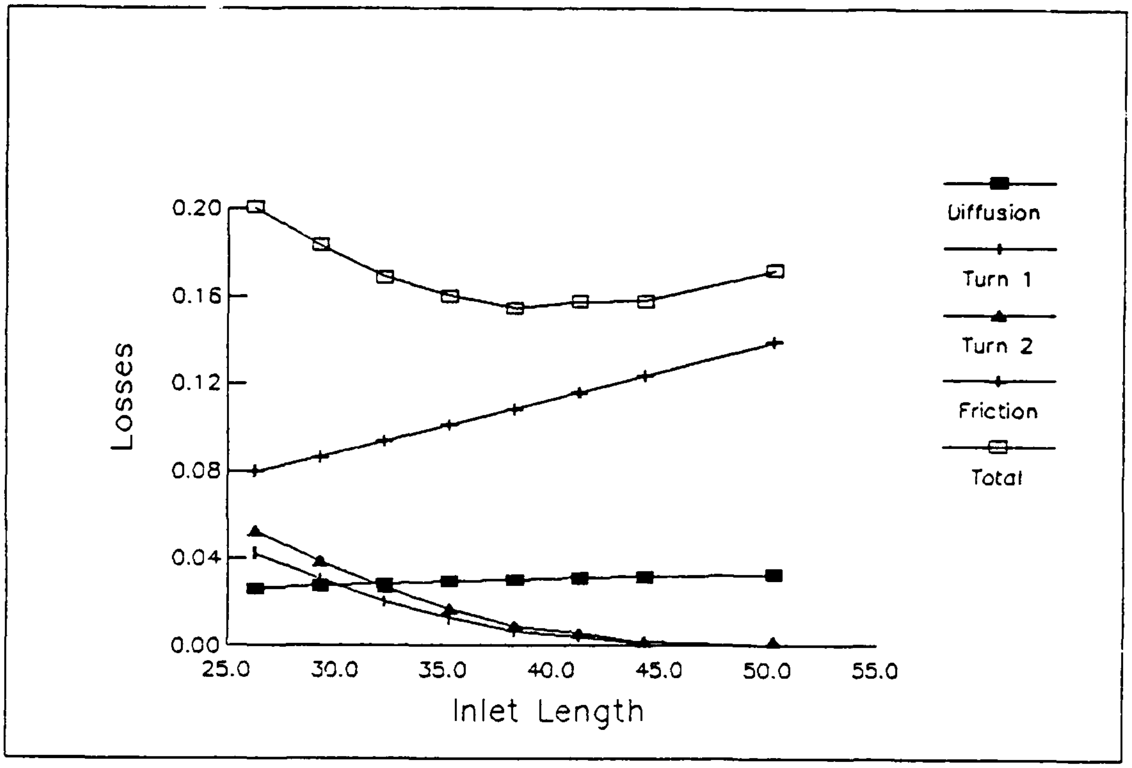


Figure 3.6. Model 320 Inlet Differential Pressures

accommodate a more complex nozzle design and the modular rail mount can handle the additional weight and loads. The only type of nozzle which the baseline design will have is a simple convergent nozzle. Variable exit geometry was not needed and the weight penalty and complexity made it undesirable.

One measure of nozzle efficiency is the pressure recovery, $\frac{P_{t_9}}{P_{t_8}}$. The pressure recovery for the typical nozzle ranges from 0.95 to 0.98 depending upon flight conditions and engine speed. At sea level static conditions, 0.95 is a typical value and was used to calculate the installed thrust.

3.2.5 Installed Thrust The engine manufacturer provided data for uninstalled performance using Mil-E-5007D recoveries [72]. This thrust data used a value of $\eta = 1.0$ for subsonic Mach numbers. Because the aircraft's inlet does not have a perfect pressure recovery, this factor must be compensated for. The installed thrust was needed to get a better estimate of vehicle performance so

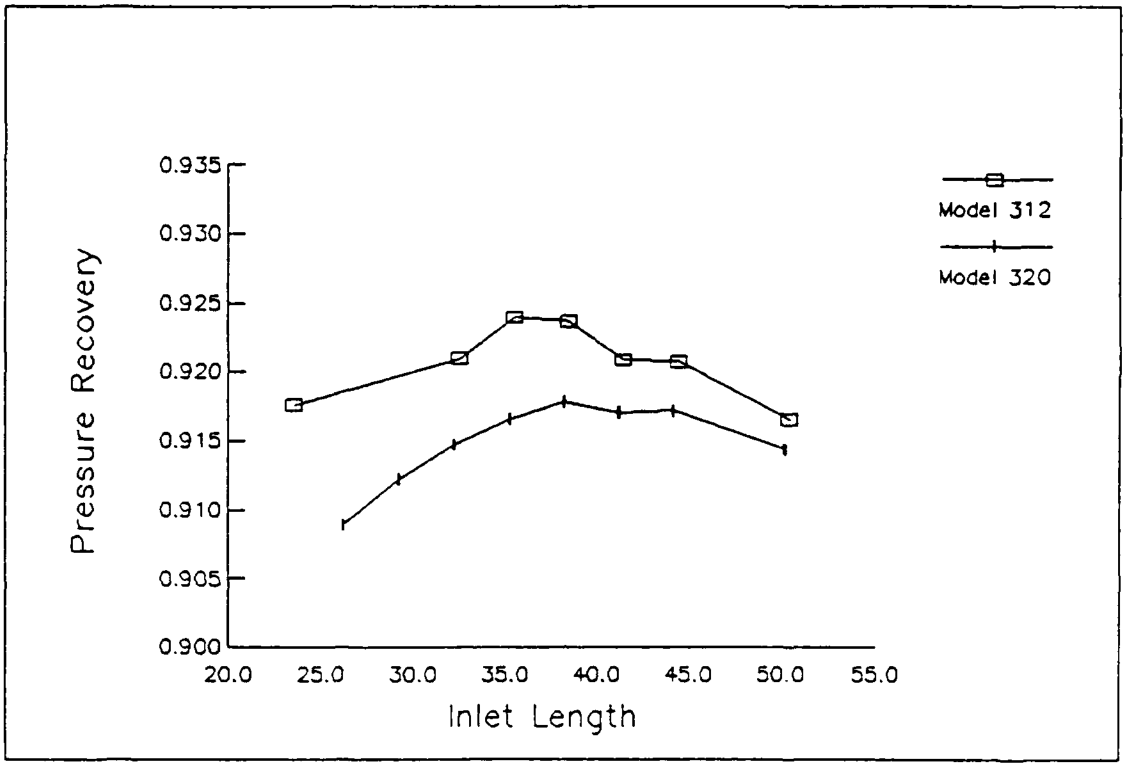


Figure 3.7. Inlet Optimization Pressure Recoveries

the different configurations could be compared. The inlet pressure recoveries for the inlets designed in Section 3.2.2 were used to calculate the installed thrust using the following equation:

$$F_n = F_g \left[1.0 - 1.5 \left(1.0 - \frac{P_{t_2}}{P_{t_0}} \right) \right] \left(\frac{P_{t_6}}{P_{t_6}} \right) \quad (3.5)$$

where

F_n = net installed thrust

F_g = gross thrust

The value of 1.5 in Equation 3.5 was determined empirically at Aeronautical Systems Division/ENFTA and represents engine thrust sensitivity to inlet efficiency. The installed thrust values for the preliminary inlets listed in Appendix J were used for the configuration selection studies in Chapter XI.

3.3 Engine Recommendation

The engine selection process narrowed the type of engine to a turbojet in the first iteration. The five candidate engines were evaluated using the analytical hierarchy process in Section 3.1.2.1 and narrowed to the two Teledyne alternatives. The inlet design was done in parallel for both engines to find the installed thrust to better estimate the aircraft performance. The installed thrust data for the two engines was used during the aircraft performance estimation in Section 11.6 to determine which engine would be better suited to perform the intended family of experiments. The Teledyne Model 312 engine led to a smaller and lighter aircraft, while the Model 320 had a superior thrust to weight ratio. The recommendation from the aircraft configuration group was strongly in favor of the Model 320. The Model 320 engine also has better specific fuel consumption and can provide more electrical power than the Model 312. Based upon the configuration group's recommendation in Section 11.6 and the additional positive attributes of the engine, the Model 320 was recommended to be carried into the detailed design phase of the project. The Model 320 better meets the overall system objectives of maximizing test versatility through a high aircraft thrust to weight ratio. The next section to be covered will be the required engine instrumentation.

3.4 Engine Instrumentation

As a test vehicle, the MURV has a substantial need for data gathering capability. The engine will be custom instrumented to meet the engine data requirements. Sensors will be installed on the engine to provide the following data:

1. Engine RPM
2. Exhaust gas temperature (EGT)
3. Exhaust gas pressure
4. Inlet temperature

5. Inlet pressure

These sensors will be sampled by the data acquisition system and recorded for analysis. The engine RPM and Inlet temperature will also be used by the digital throttle controller. EGT and RPM will also be monitored by the pilot at the ground station.

3.5 Fuel System Design

The fuel control system for the engine will be supplied by engine manufacturer. The Teledyne Model 320 control unit is a digital system with a control input from the flight control system. The largest component of the fuel system is the fuel tank and this will be addressed next.

3.5.1 Fuel Tank design The Fuel tank needs to:

1. be compact in size by holding the required capacity of fuel in the minimum volume,
2. be lightweight, and
3. minimize the fire danger in the event of a crash.

The objectives and Measures of effectiveness which were used for evaluation of candidate fuel tanks followed directly from these subsystem needs and are listed below:

1. To minimize size, measured by the overall length because the height and width are constrained by the fuselage structure.
2. To minimize weight, measured by the weight of the tank.
3. To minimize fire danger, measured by the type of fire prevention measures of the tank.

These needs and objectives helped to guide the development of candidate fuel tanks which are discussed next.

To reduce the shift of the center of gravity during fuel consumption the fuel tank was located near the aircraft's center of gravity. This caused the rear of the tank to be placed over the engine inlet duct for some of the preliminary configurations. One remedy for this conflict was to form the rear of the tank to fit over the inlet. By adapting the fuel tank in this way, the tank was longer than it had to be, but it increased the modularity of the system because the tank could be placed in more locations than a rectangular tank without the cutout. The fuel tank will have to be specially constructed for the MURV, so the impact on the design will be minimal. The constraint on the fuel tank was that it had to fit between the aircraft's structural mounting rails during installation and removal. For the preliminary design, the fuel tank was sized to hold the approximate amount of fuel required to run the engine at full throttle for a mission duration of 25 minutes.

The candidate solutions were broken down into two classes, rigid and flexible. The rigid tanks were made out of either metal or composite materials and the flexible tanks were made out of neoprene rubber. The metal tanks were very similar to the composite tanks but had a higher weight. To prevent rupture upon impact, the rigid tank could be lined with a bladder; and to help prevent risk of fire, either type of tank could be filled with a low density open foam. The foam has the disadvantage of displacing a small amount of fuel which increased the size of the tank to get the required volume of fuel. The rigid tanks were rectangular and sized to fit between the structural rails. The flexible tank was shaped to fit with the sides flush against the aircraft side walls with cutouts for the structural rails. During installation the flexible tank would be squeezed between the rails and into place. With the fuselage supporting the sides and bulkheads in front and rear, the tank would be mounted securely in place. The candidate solutions and their advantages and disadvantages are listed below in Table 3.10. The flexible neoprene tank best fit the objectives for the fuel tank because of the advantages listed in the table. The small weight penalty was compensated by the safety features of the tank. This type of flexible tank is used for other RPV's and for racing car fuel systems and should work well for the MURV. The preliminary fuel tank design is shown in Figure 3.8. The fuel supply system will be discussed in the next section.

Table 3.10. Candidate Fuel Tank Advantages and Disadvantages

Candidate Tank	Advantages	Disadvantages
Basic rigid	lowest weight of the rigid tanks	danger of rupture danger of fire basic size
Rigid with bladder	low danger of rupture	higher weight basic size danger of fire
Rigid with bladder and foam	low danger of rupture low danger of fire	highest rigid weight size due to shape and foam
Basic flexible	low danger of rupture compact size	no structural shape danger of fire
Flexible with foam	low danger of rupture low danger of fire compact size	higher weight

3.5.2 Fuel Supply System An advantage of the Teledyne engine is the integral fuel pump, so no external pump is required. A small pressure of ten psi is needed to initially start the engine. This pressure will be supplied by a small compressor bleed of approximately 0.10 percent which will not affect the engine performance [12]. During engine start, the fuel system will be pressurized by the compressor bleed as the engine is spooled up. The fuel will be routed through a fuel flow meter to the engine fuel control valve to the engine. The fuel flow meter is an added sensor to keep track of the fuel on board the aircraft. Its output will be sampled by the data acquisition system and telemetered to the ground.

An in-flight fuel dumping system was considered to increase safety by minimizing the danger of explosion during a crash or forced landing. This system would be composed of a drain line from the fuel tank through a solenoid valve to an overboard nozzle. Possible locations for the nozzle would be the bottom of the fuselage, out of the boattail under the tailpipe, or out of a port on the wing. The dumping valve would be controlled by the flight control system and activated from the ground. The advantages of having a dumping system were weighed against the disadvantages. The advantages of the system were that excess the fuel could be disposed of in a emergency situation to reduce the risk of explosion and fire, and the landing weight of the aircraft could be

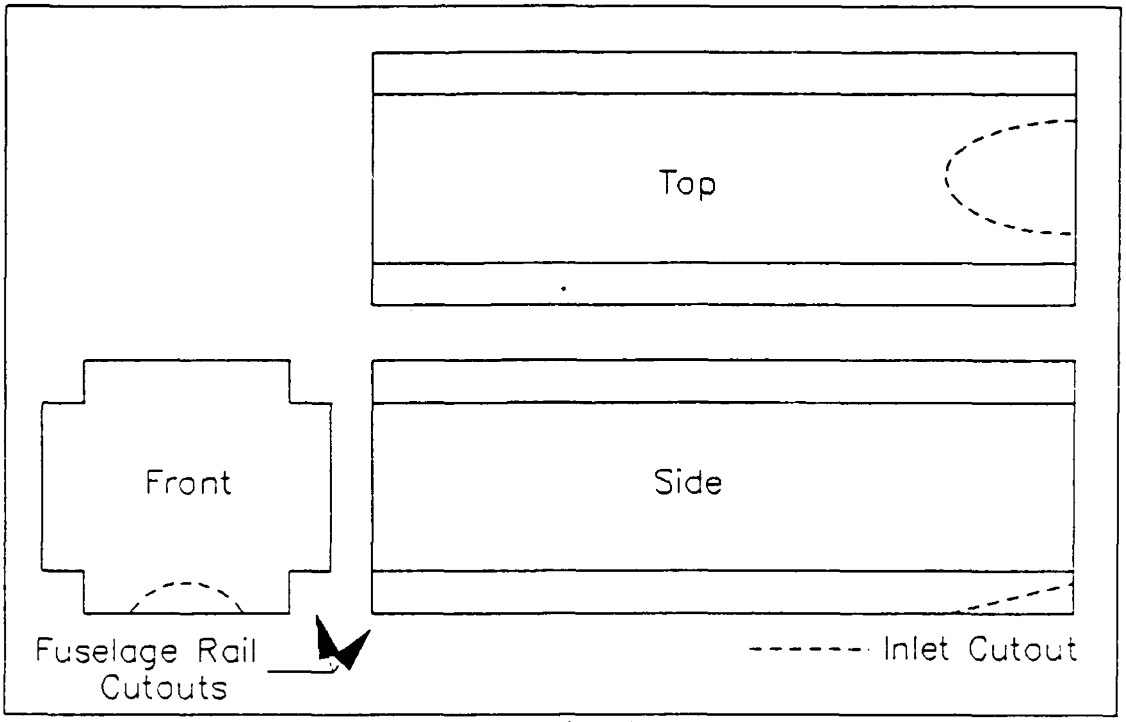


Figure 3.8. Fuel Tank Design

substantially reduced during an aborted mission. The disadvantages of the system include the increased complexity of the fuel system with another possible failure mode, the increased weight of the system, and the inherent risk of igniting the dumped fuel. Another disadvantage of a dumping system was the decrease in modularity. For example, when the aircraft is reconfigured with a new wing, the fuel system may have to be moved to compensate for a shift in the center of gravity. The movement of the fuel system would be more complex because of the additional mounting provisions needed for the valve, hose and nozzle. Location of the nozzle on the wing would add unnecessary complications when removing or modifying the wing, but would isolate the raw fuel from the jet exhaust. The disadvantages of a fuel dumping system were judged to outweigh the two advantages gained by having the system, so the in-flight dumping system was not included as a part of the baseline design.

3.6 Electrical Power System

The MURV has a large electrical power demand to operate the flight control system and the data acquisition system. The power requirements for the flight control system were driven by the processing unit and actuators. The data acquisition system requires power to operate the telemetry receiver, transmitter, processing unit, sensors, and video camera. The Teledyne engine will be fitted with an alternator to supply the main source of electrical power [12]. Additional power needed will be supplied by the auxiliary power system. A back up electrical power system should be available to fly the aircraft in the event of an engine or alternator failure. The amount of power required from the auxiliary system will depend upon the alternator output from the engine selected in the detailed design phase and the requirements of the actual subsystem components. The auxiliary power system could be composed of batteries or fuel cells. The fuel cell was dropped from consideration because of the size and complexity required for use. Primary or secondary batteries could used. Primary or expendable batteries are non-rechargable and cheaper than rechargable secondary batteries. A common primary battery is made of carbon-zinc cells which produce between 29 and 35 watt hours per pound (whr/lb). From a maintainability standpoint, secondary rechargable battery is preferable. Nickel-cadmium batteries produce between 8 and 20 whr/lb and to supply an additional 1.5 KVA for one half hour, typical nickel-cadmium batteries weigh between 40 and 90 pounds [49:1534-47]. The weight of the batteries needed to produce this amount of electricity was one of the reasons the alternator output from the engine was given a high weighting during the engine selection process in Section 3.1.2. Electrical power requirement estimates will not be available until the detailed design phase when specific actuators and other subsystem components are selected. The type of battery to use depends on the discharge rate, capacity and reusability of the battery system. This decision can not be made until estimates are available for the power usage during the detailed design. A list of battery types and power output is listed in Appendix K. The next major subsystem to be covered will be the engine starting system.

3.7 Engine Starting System Components

The Teledyne turbojet engine can start by firing a cartridge or through the use of high pressure impingement air on the compressor [12]. The engine will be configured to use one or the other starting systems. Cartridge starting is commonly used on missiles which are stored for long periods and then fired on short notice. The advantages of using a cartridge system is the self contained nature of the starting process. The system is fast and reliable with the engine going from 0–89,000 RPM in four seconds [12]. The pilot has only to flip a switch on the ground station to ignite the engine. The disadvantages of this system are that the risks involved using a pyrotechnic charge, the uncontrollability of the engine start, the weight of having the start cartridge on the engine, and the cost of using an expendable item for the starting process. The impingement starting system is used by most jet aircraft engines and during engine development testing. The advantages of this system are the simplicity and safety of using compressed air to start the engine turning, the starting process is controllable and can be aborted at any time, and the cost per start is small. The air can be supplied by a high pressure bottle which can be refilled. The disadvantages are that an air hose has to be connected to the fuselage during starting. This can be done with a small access panel in the engine compartment with a quick disconnect air fitting. The impingement air system was chosen for use with the MURV because of the simplicity, lower weight, and other advantages listed above.

3.8 Propulsion System Summary

The prelimininary propulsion system design has covered engine selection and the areas essential for engine integration. The engine selection narrowed the choices of engines to the two Teledyne turbojets from the range of jet-like thrust producing engines. Inlet design, analysis, and optimization were covered as part of the engine integration. The preliminary inlets were designed for the two engines, and the installed thrust was calculated and used during the external configuration design. The final engine recommendation of the Teledyne Model 320 was determined based upon the aircraft configuration studies and the attributes of the engines. Other major subsystems were discussed ranging from the fuel system to the power systems.

IV. Flight Control System Development

This chapter describes the development of the flight control system for the MURV. The design approach used is discussed in the context of the special attributes of a high performance remotely piloted vehicle. These attributes are used to define general requirements for the MURV.

The MURV will be a multi-purpose remotely piloted flight vehicle providing capabilities spanning over large range of experiments. Since this will be achieved by physically modifying the flight vehicle, corresponding adjustments to the flight control system will be required. Accommodating this variability is the challenge for the control system designer. The variations in aerodynamic properties and MURV missions will require a flexible flight control system. MURV requirements were derived from the family of experiments defined in Volume One, Chapter Four and alternative configurations identified in Volume Two, Chapter One.

4.1 Design Objectives

Objectives of the flight control system development were derived from the top level objectives from Volume One, Chapter Two. Versatility of the flight control system is an integral part of the overall modularity of the MURV system. A majority of the proposed experiments, terrain following, maneuvering and integrated controls are conducted in conjunction with a unique flight control system. Some experiments will require particular control surfaces and subsystems, while most classes of experiments will require mission specific control algorithms. Within each class there will be a variety of flight modes for each configuration. This multi-mode capability adds breadth to the research which can be conducted with each configuration.

For low risk operation adequate control must be provided by the control system. Since an objective of the vehicle configuration development is to provide highly maneuverable designs some, vehicles may have be neutrally stable during some mission phases. In this case, the flight control system must provide stability augmentation for controlled flight. In addition, the MURV is remotely piloted and, as such, the flight control system should tolerate pilot inefficiencies. This can be addressed via handling quality specifications and flight control system stability augmentation.

Since airborne failures can be catastrophic, the MURV should have strict reliability requirements. All flight critical elements will have at least one level of protection. Flight critical elements of the flight control system include:

- Flight computers
- Actuators
- Power distribution unit
- Telemetry links
- Sensors
- Other avionics

Levels of redundancy and component reliability factor into the overall MURV system reliability. The MURV should have redundant flight critical elements, wherever possible, to ensure safe operation after component failures.

The control system should also minimize interface requirements with other subsystems. This objective is directly related to the goal of maximizing the modularity of the system. The control system requirements will be applicable to the range of MURV vehicle configurations.

Finally, the MURV system should be designed with cost as a factor. Emphasis should be made on affordability in the absence of a hard constraint. Feasible designs were compared in terms of relative cost.

The design objectives for the MURV control system design can be summarized as:

1. Provide mission flexibility
2. Provide adequate control for defined missions

3. Provide adequate reliability

4. Minimize constraints on functional interfaces

5. Be affordable

4.2 Design Approach

There are many approaches to the design of a flight vehicles control system. The method stated below is an example of one such approach. At this point in the MURV design process, the identification of fixed vehicle characteristics was not complete. The mission requirements and vehicle characteristics are variable elements in the process.

1. Definition of mission requirements
2. Identification of fixed system characteristics
   - Vehicle configuration
   - Subsystem components
   - Environmental conditions
3. Development of feasible designs/functional diagrams
4. Systems analyses and design selection
5. Detailed design

Our goal was to develop a conceptual design of the flight control system which addresses the unique requirements of the MURV. Several aspects of a remotely piloted vehicle, designed for high performance, need to be considered. Since the MURV is remotely piloted, a remote cockpit is introduced in the control loop via telemetry. This will increase the system time delay in the control loop and introduce additional reliability factors. A remote cockpit also deprives the pilot of beneficial physical cues. As an experimental aircraft, the MURV will be capable of high performance maneuvers and acquire data for analyses. The timely acquisition of data is important for both safe flight operations and the success of the experiment. In addition, the MURV will require some form of automatic control for preplanned flight maneuvers and backup control. Preplanned flight maneuvers provide controlled test conditions for experimental data acquisition. Some form of backup flight control is necessary to provide safe operational control if the primary control loop fails. These failures are addressed later in this chapter. Finally, on-board weight, volume and power requirements are always tightly constrained on aircraft and even more so on RPVs. Where possible functional elements should be allocated to the ground segment of the MURV. These are the important factors in the MURV control system design problem. They were used to develop feasible system designs. For the conceptual design of the flight control system our goal was to:

1. Select specific mission scenarios
2. Develop control system functional requirements
3. Identify/analyse feasible designs
4. Develop interface requirements

Current control system designers begin the process at the earliest stage aircraft development. The flight vehicle's configuration and its associated aerodynamic characteristics are the largest influences on aircraft performance. Cooperation between the control system designer and the vehicle designers can influence the configuration development to ensure the attainability of specific mission requirements.

4.2.1 Mission Scenarios The MURV will be capable of the conventional flight phases require for standard operations. Requirements beyond these depend on the experimental mission. Since supermaneuverability was identified as a primary design driver, possible flight modes for these tests were identified. Uncoupled flight modes provide high maneuverability and have been demonstrated by the AFTI F-16 test program. As discussed in Volume One, Chapter Three, the Herbst maneuver shows some promise in this area. These advanced control concepts are representative of those which could be developed and implemented by AFIT researchers. The MURV will be capable of safely operating in the flight phases and modes listed below.

- Conventional flight phases
  - Take-off
  - Climb
  - Cruise
  - Standard turn
  - Landing
- Maneuver flight modes
  - Pitch Pointing
  - Vertical Translation
  - Direct Lift
  - Yaw Pointing
  - Lateral Translation
  - Direct Sideforce
  - Herbst Maneuver

Using the stated objectives and the identified MURV flight mode requirements a choice was made to select digital control over an analog system. A digital system is more flexible in terms of both design and implementation. To perform the specified missions and provide the modularity desired, a digital flight control system is a superior approach.

The flight modes also impart requirements for particular vehicle functional elements. Functional requirements for sensors and flight control surfaces were derived from these particular flight modes, with the exception of the Herbst maneuver. These requirements are addressed in the following sections. The Herbst maneuver requires thrust vector control which was not implemented in the preliminary design.

4.2.2 Development of Requirements Since the key factors of the MURV design are undefined at this stage of the design, the functional requirements for measurements and control surfaces were derived from MURV flight modes. The use of the uncoupled flight modes for this purpose was appropriate for developing preliminary requirements. The specification of the architecture of the control system was influenced by the remote pilot requirement and factors such as the partitioning of tasks, data distribution and failure protection.

4.3 Measurement and Discrete Data Requirements

Proportional measures of vehicle states and flight conditions provide the control system with the information it requires to track command inputs and generate experimental data for analysis. The number, rate and resolution of data items needed to support the control system are directly related to specifications for the flight test instrumentation system.

4.3.1 Required Data Items The on-board data requirements were identified by reviewing the possible MURV flight modes and phases [89]. The remote cockpit also provides proportional input, e.g. stick positions, to the control system These parameters are considered preliminary. The resulting data list is the union of the requirements for each mode. Table 4.3.1 summarizes the survey of flight mode requirements.

The following parameters are required to support the MURV's baseline requirement for the multi-mode control system.

Table 4.1. Onboard Measurement Requirements

Flight Mode	Required Measures
Pitch Pointing and Vertical Translation	Flight Path Angle Pitch Rate Normal Acceleration Elevator Deflection Flaperon Deflection
Direct Lift	Pitch Rate Angle of Attack Elevator Deflection Flaperon Deflection Normal Acceleration
Yaw Pointing and Lateral Translation	Heading Lateral Direction FPA Roll Rate
Direct Sideforce	Yaw Rate Bank Angle Roll Rate Yaw Rate Lateral Acceleration

- Onboard data
  - Pitch attitude
  - Pitch,roll and yaw rates
  - Normal and lateral acceleration
  - Dynamic pressure
  - Static pressure
  - Angle-of-attack
  - Angle of sideslip
  - Control surface deflections
  - Engine RPM
  - Engine inlet and exhaust pressures

- engine inlet and exhaust temperatures
- Engine thrust vector direction (not in baseline)
- Fuel flow rate

- Remote cockpit
  - Stick pitch and roll forces/positions
  - Stick rudder force/position
  - Throttle position

While these proportional parameters provide data for particular control laws, discrete information from the remote cockpit should be used to select or activate particular flight control system tasks. These tasks could be:

1. Flight mode selection - implement specific flight control law
2. Built-in-test - conduct control system diagnostic test
3. Engage back up control system - pilot initiated switch
4. Preplanned flight maneuver - execute preprogrammed flight plan

In addition to the aforementioned data, status of flight critical hardware should be monitored on-board and transmitted via telemetry to the pilot station. Approximately twenty four proportional parameters and an undefined number of discrete commands will be required for the flight control system. The number of discrete status flags will be determined when the final design of MURV subsystems is complete.

The MURV should have reliable data for both the primary and backup control loops. Flight critical sensors should be redundant. For the preliminary design, these are defined as those sensors which are required to support a simple controller.

4.3.2 Data Rates Some MURV configurations will be statically unstable. In this situation the control system should maintain stable flight. Since there are telemetry links and a remote cockpit in the control loop, the control cycle time is increased. In combination, these factors will lead the designer to reduce the sampling time. Sampling frequencies from similar, existing systems provide "rough order of magnitude" estimates. Similarity, in this context, simply means that the control systems considered support high performance control functions such as relaxed static stability and/or were remotely implemented. The range of sampling frequencies surveyed was up to 220 Hz with command rates on the order of 40 - 55 Hz [19,104] and [55]. While these are only estimates, they indicate that sampling time should an important design factor.

4.3.3 Measurement Precision The analysis of finite word length effects begins with the selection of a candidate length. For the purpose of the design 12-bit word resolution was selected. This is compatible with 16-bit and 32-bit microprocessors and 16-bit data bus architectures. For a 16-bit architecture and a 12-bit data word length at least two bits will available for discrete data.

4.3.4 Interface Considerations Each particular flight mode has a set of required state information needed for implementation. Control applications not considered in this design may require the addition of some states. The goal in the preliminary design was to identify foreseeable requirements and include them in the baseline. Discrete function requirements will also vary. The addition of functions, such as, air brakes, two position forebody strakes and a parachute introduce requirements in both the telemetry system and the remote cockpit. As the MURV evolves into different configuration these changes should be identified as early as possible. This should provide time for the incorporation of these requirements.

4.4 Flight Control Surface Requirements

Changes in aircraft direction are imposed by force generated via control surfaces. The MURV requires control surfaces which can provide independent state control. Table 4.4 identifies the control surface combinations required for uncoupled flight path control. The specific combinations required for the MURV should be determined in the detailed design.

Table 4.2. Control Surface Requirements

Flight Mode	Required Control
Pitch Pointing and Vertical Translation	Elevators Flaperons
Direct Lift	Elevators Flaperons
Yaw Pointing and Lateral Translation	Rudder Aileron Canard
Direct Sideforce	Rudder Ailerons Canard

Control surfaces which could be required for conventional flight phases or other flight modes are:

1. Leading edge flaps - provides increased wing lift
2. Trailing edge flaps - provides increased wing lift
3. Dorsal fins - provides directional stability at medium angles of attack
4. Ventral fins - provides directional control at high angles of attack
5. Forebody strakes - provides increased yaw effectiveness at high angles of attack

The selection of control surface actuators requires the specification of three parameters: maximum hinge moment, maximum deflection and maximum surface rates [63]. The maximum hinge moment typically occurs at low altitude, at a high dynamic pressure. The maximum deflection is determined by the moments produced by the control surface at lower altitudes. The maximum surface rates are also set at lower altitudes and should, as a minimum, satisfy the stability and control requirements and achieve the responses specified in MIL-F-8785C. See Volume Two, Section 6.4 for thoughts on implementation.

Control surface reliability was considered. The MURV-320 configuration design does not specify control surfaces. At this point of the design the, flight critical surfaces have not been identified. Providing adequate control of the vehicle in the event of a failed surface can be handled two ways. The first requires the use of redundant actuators (duplex or triplex) in the control loop. Each "side" of the actuators would be accessible to either the primary or backup control system. The other approach would be to "split" the control surfaces and use a single or simplex actuator driven by a reconfigurable control law. The use of redundant actuators is the more proven method and is recommended for the baseline design.

There are two possible concepts for the MURV actuators, hydraulic powered and all-electric actuators. There are pros and cons associated with each option. Hydraulic systems (3000 psi) are technologically mature and lend themselves to design reliability. On the other hand, they tend to be heavier than their all-electric compliments. While electro-mechanical actuators themselves weigh more, the overall system weight is significantly less [63]. All-electric actuators are considered more energy efficient but may require a high voltage power source. Current designs utilize 28 - 270 volts dc. High performance all-electric actuators are currently in development as are 8000 psi hydraulic systems [63]. The selection of the actuation system was deferred to detailed design. However, an all-electric actuation system improves the modularity of the MURV and would reduce total flight control system weight. Therefore, if all-electric actuators provide the desired performance with acceptable power consumption, they will be implemented in the MURV. Table 4.4 contains the rate and range of the hydraulic powered control surfaces employed on NASA's HiMat remotely piloted research vehicle [55]. These specifications may be indicative of high performance control surface requirements and could be used to identify the range of these parameters in future design iterations.

4.5 Flight Control System Architecture

Our goal was to identify feasible control system architectures and determine which was the most appropriate. The flight control system architecture can be broken down into three areas:

Table 4.3. HiMat Actuator Parameters

Control Surface	Rate (deg/s)	Maximum Deflections (degs)
Elevators	76.6	+28 to -21
Elevons	87.2	+27.5 to -20
Rudder	65.6	+10 to -10
Ailerons	86.8	+20 to -20
Canard	87.3	+18 to -20

functional partitioning, data distribution and failure protection. Feasible designs are presented and evaluated based on the following design objectives:

1. Provide mission flexibility
2. Provide adequate control for defined missions
3. Provide adequate reliability
4. Minimize constraints on functional interfaces
5. Be affordable

4.5.1 Elements of the Flight Control System The control system central computer provides the computational power to process the input data via resident control laws and provides a command vector to achieve the desired aircraft response. Other microprocessors are required to manage such functions as servo-actuator control, failure detection, uplink and downlink data processing.

In the absence of detailed specifications, we recommend the 32-bit class of microprocessors for the MURV system. This type of microprocessor is becoming an affordable controller for high performance control applications [66]. The growth capability of the MURV should be enhanced by the specification of a 32-bit microprocessor. The 32-bit machine is powerful and its full capability may not be required initially, but the flexibility it offers to the control system designers satisfies a top level objective.

The central flight control computer is flight critical element in the DFCS. Some provision should be made to maintain control of the MURV in the event the central computer fails. In full-scale aircraft redundancy is provided using up to four independent control systems composed of quadplex sensor sensors and triplex actuators. For the MURV this approach would not be desirable. So, as in the case for flight critical sensors, there should be some reduced capability alternative which provides for safe MURV control. This should be accomplished with a backup controller on-board the MURV. This would provide the redundancy for a failed central computer and or a failed telemetry link.

4.5.2 Control Law Development The control laws implemented by the MURV depend on the experimental mission. Requirements regarding the method of control were left unspecified since the dynamic structure of the system was undefined. Performance requirements are stated in AFFDL-TR-76-125 [80]. The following list identifies the performance specifications which are available.

1. Attitude Hold (Pitch and Roll)
2. Pitch Transient Response
3. Roll Transient Response
4. Heading Hold
5. Heading Select
6. Turn Transient Response
7. Altitude Coordinated Turns
8. Altitude Hold
9. Mach Hold
10. Airspeed hold

4.5.3 Interface Considerations The are several important control system interface concerns. First the control laws must satisfy the MURV mission requirements. The flight modes required to conduct a flight test must be identified. Since the MURV will have multiple configurations and its missions vary, the development of flight control laws will an ongoing process. A good software support environment is strongly recommended.

Second, the complexity of the control law must be compatible with the the data sampling rates. The computational load on the central computer should limited so that:

$$\tau_{a/d} + \tau_{comp} + \tau_{d/a} << T$$

where $\tau_{a/d}$ and $\tau_{d/a}$ are conversion times, τ_{comp} is the computation time and T is the data sampling time. In addition, the specified word length of the flight data acquisition system should be compatible with the word length employed by interfacing microprocessors. Changes in requirements in these areas will require long lead efforts to make them compatible.

4.5.4 Functional Allocation Since the MURV is is a remotely piloted vehicle there exists some flexibility in locating the elements of the flight control system. Those which must reside on-board are:

1. Sensors
2. Signal conditioners
3. Control surface actuators
4. Servo-motor controllers
5. Onboard flight controller

The flexibility lies in the location of: 1) the central flight control computer containing the control laws, 2) an autopilot and 3) a navigation computer. Tracking commands will be generated at the remote cockpit and telemetered to the aircraft. Potential designs were developed using the design objectives as guidelines. The system designs are Figures 4.1,4.2 and 4.3.

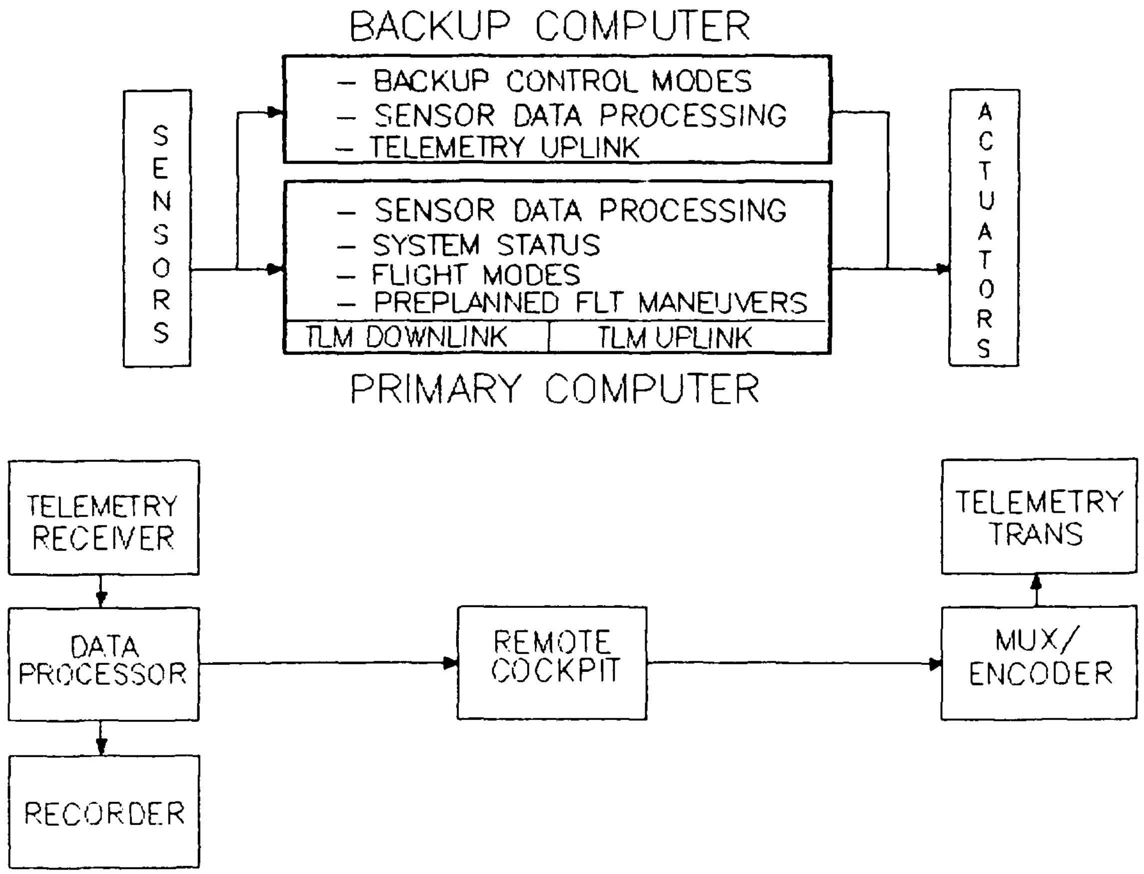


Figure 4.1. Flight Control System #1

All of the designs satisfy the top level objectives. Each can provide a multi-mode control capability and adequate control for defined missions. The reliability and integration objectives are satisfied to varying degrees and all can be considered affordable for the scope of this study. The differentiating factors between these designs are flexibility and on-board weight and volume.

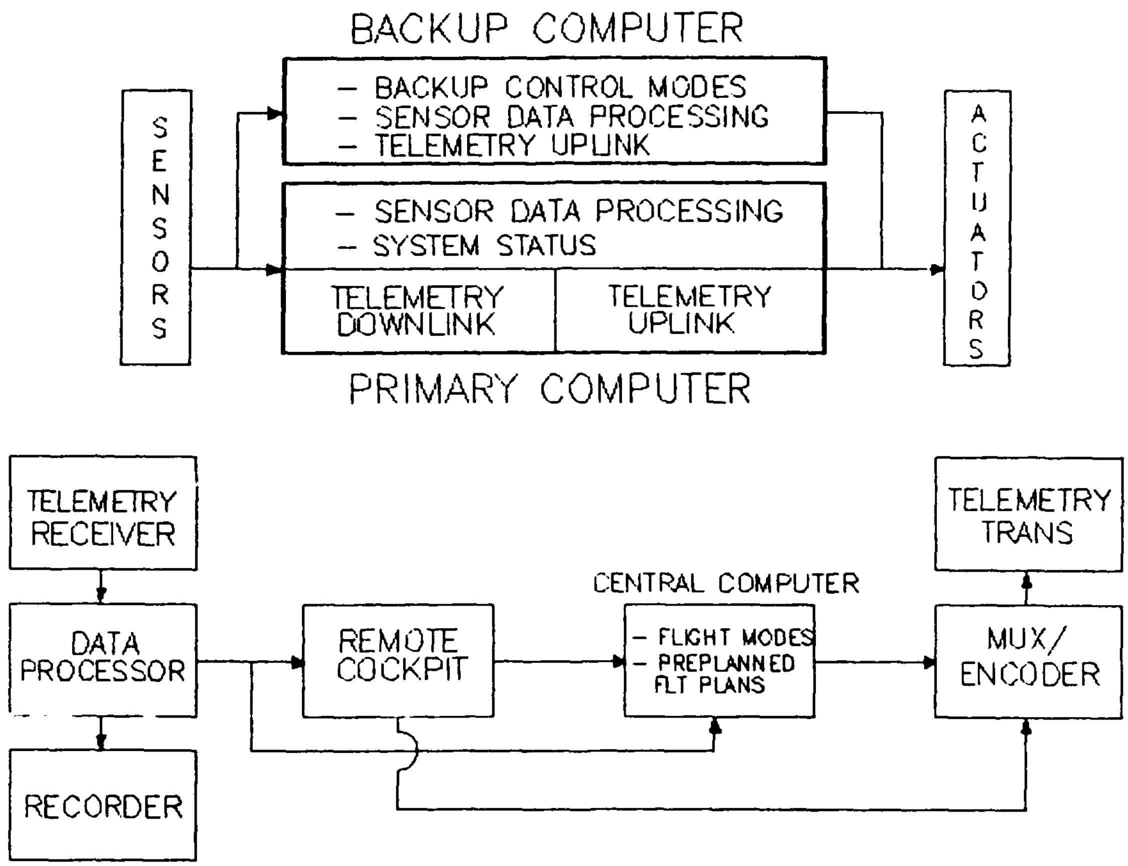


Figure 4.2. Flight Control System #2

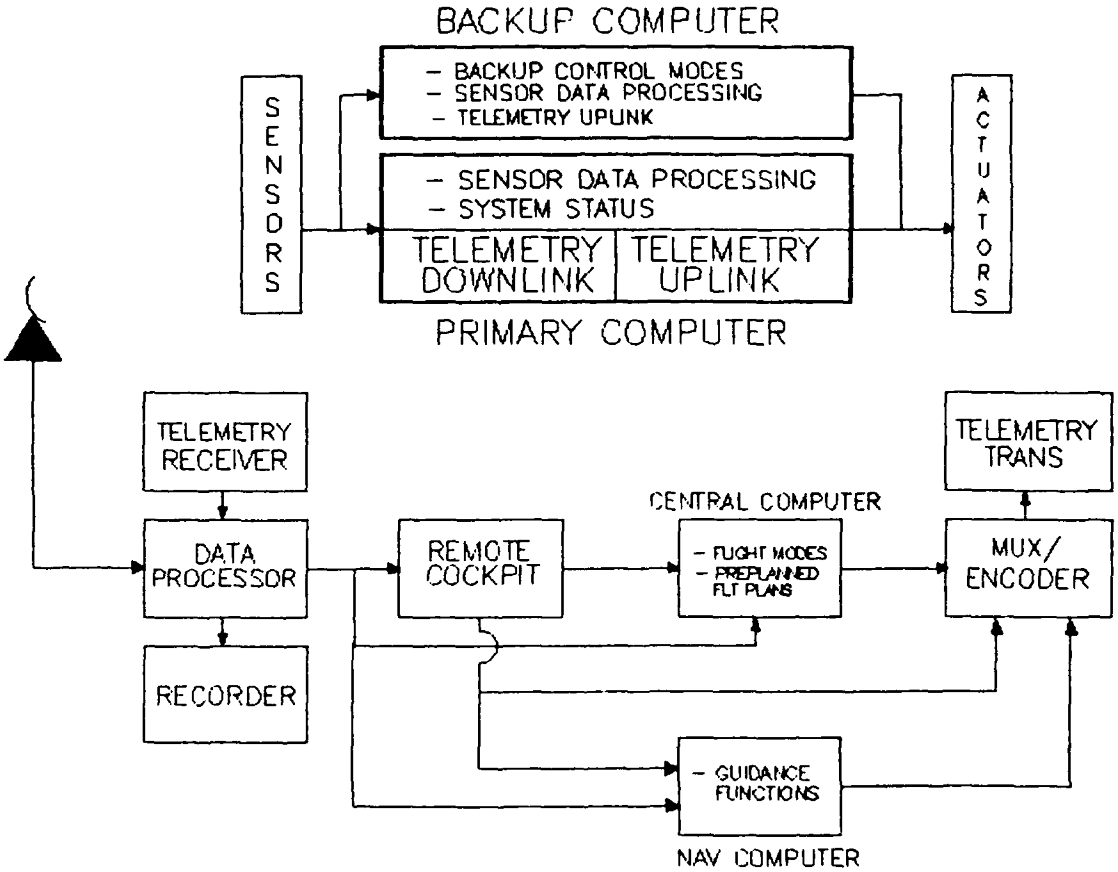


Figure 4.3. Flight Control System #3

Each of the designs provides a backup flight computer with a telemetry uplink to provide safe control if the central flight computer fails. Figures 4.4, 4.5, and 4.6 are preliminary breakdowns of the functional elements of the MURV flight control software.

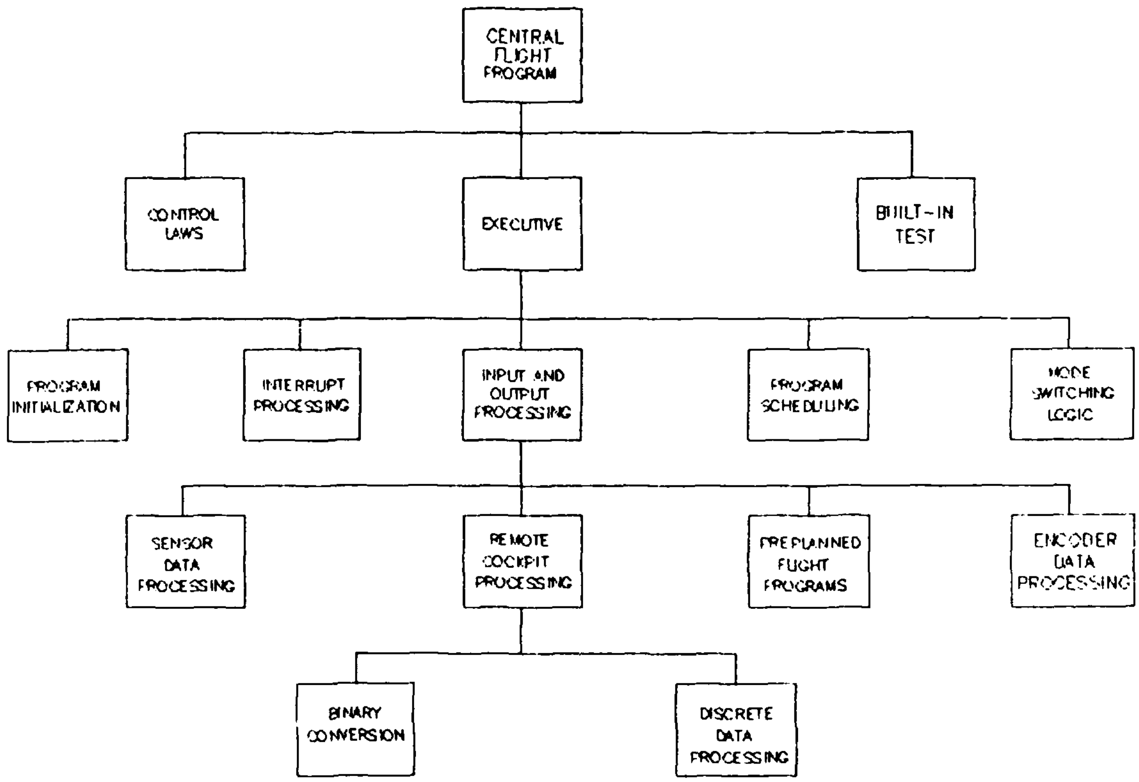


Figure 4.4. Central Flight Control Program

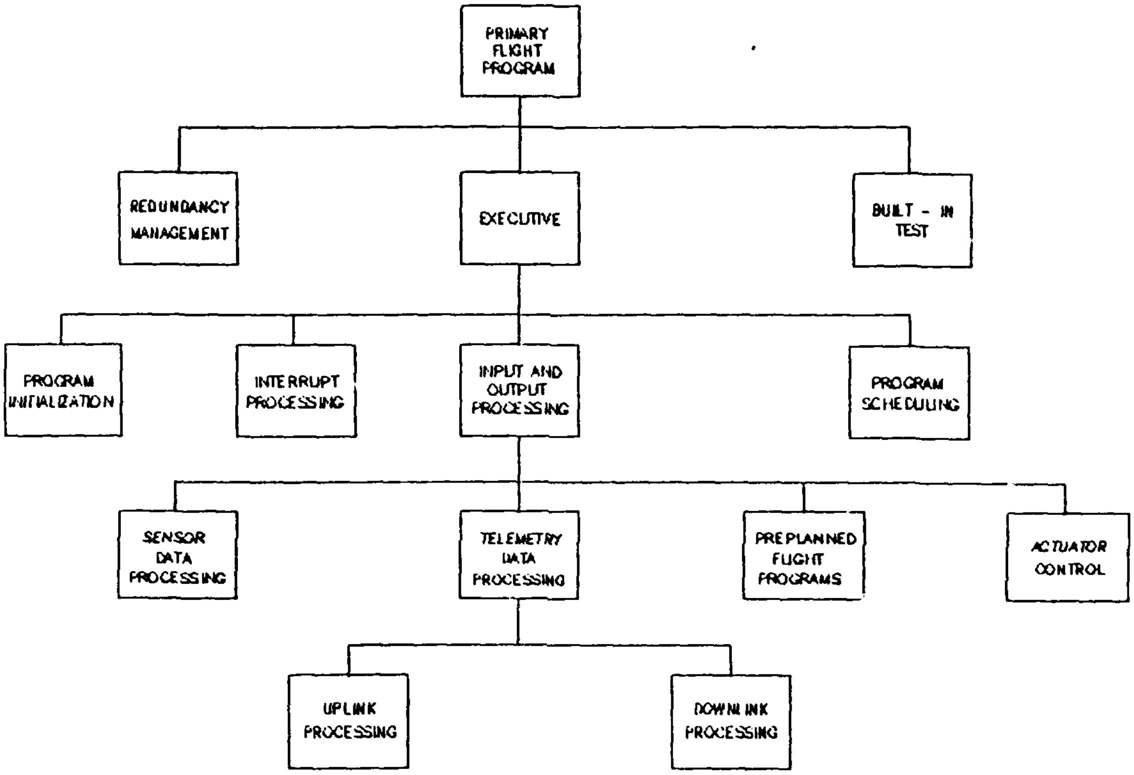


Figure 4.5. Primary Flight Control Program

The proposed designs were graded against the design objectives.

Mission Flexibility

All the designs provide multi-mode control capability. Design #1 is the most restrictive since all control laws and preplanned flight maneuvers would reside on-board. This constraint would possibly diminish the number of available flight modes for the pilot. Designs #2 and #3, on the other hand, could accommodate any reasonable number of flight modes. The AFTI-F16

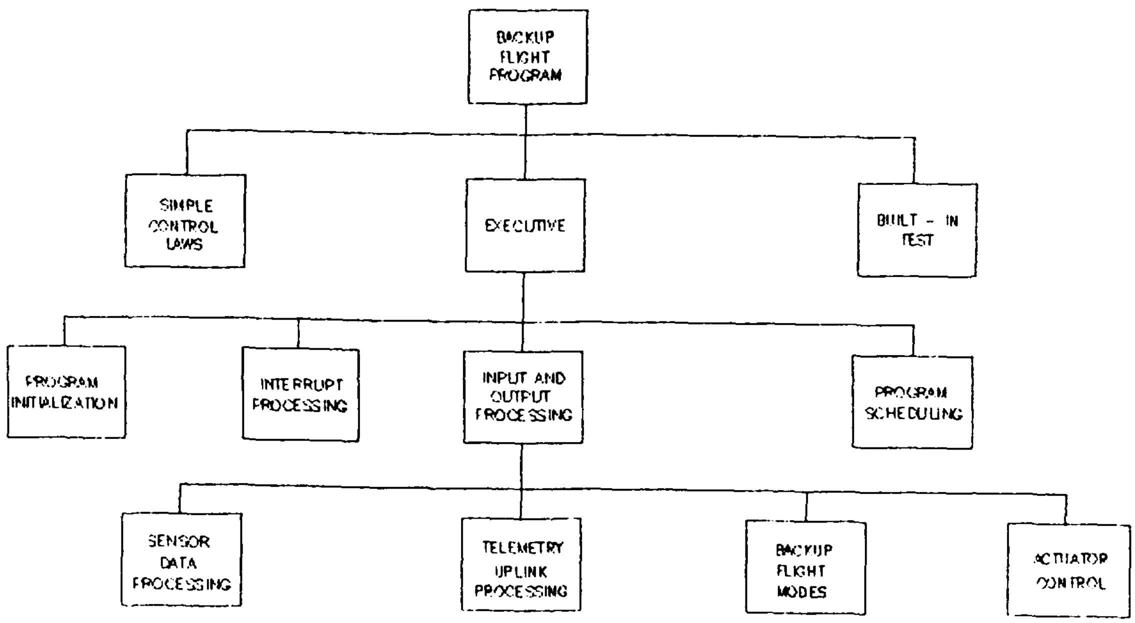


Figure 4.6. Backup Flight Control Program

test program utilized as many as eight control modes. A ground based central computer design would also reduce the control law efficiency requirements allowing the researcher to focus on the design principle as opposed to satisfying implementation constraints. Design #3 provides the most flexibility since it provides a navigation function for possible experiments.

Adequate Control

Each design will be capable of providing adequate control. Design #1 has the advantage of an on-board central processor. This would eliminate the downlink transmission delay and reduce the required sampling frequency. The ground based central computer requires higher sampling frequencies to provide current state information to ensure stable operation. Adequate control would be easier to implement with Design #1.

Reliability

The reliability of the resulting designs can be qualitatively compared by the number of functional elements in the design. Design #2 incorporates a ground based central computer into Design #1. This additional element and its associated reliability factors will reduce the overall reliability of the system. The same argument holds for Design #3 where a navigation computer is introduced to the system. Design #1 is probably the most reliable.

Interface Requirements

The flight control system is integrated with practically every other subsystem. The level of interaction can be reduced by increasing the autonomy of each functional element. Design #1 requires only the remote cockpit and the telemetry uplink to provide a control command vector. Design #2, in addition, requires the telemetry downlink to provide the state information to the ground central computer. Design #3 could possibly require a tracking system to provide an independent position estimate.

Flight control system on-board weight, volume and power requirements are also interface considerations. Designs #2 and #3 are equal in these categories to the resolution of this design.

Design #1 would increase all of these parameters. Allocating the control laws and preplanned flight maneuvers on-board would increase the computational burden which would require more circuit card, more chassis volume and electrical power. Design #2 would have the minimum interface requirements.

Affordability

The design progressively increase in cost. Design #2 requires a ground computer to compute the control laws and house the preplanned flight maneuvers. Design #3 adds a navigation computer and possibly some type of tracking system. Therefore, Design #1 in the least costly of the three designs.

Design #2 was selected because it provides the flexibility required and off loads some of the functional requirements to a ground computer saving weight, volume and power. This design is better suited to the design and testing of advanced control concepts. Design #3 has these features but also adds interface requirements and cost. The supermaneuverability experiments do not require this capability and as required the navigation function could be implemented at some future time.

4.6 Development Summary

The selected architecture, Figure 4.2, locates the central flight computer on the ground and a primary and backup computer on-board the vehicle. The central computer program residing in the ground computer will contain the MURV flight control laws and preplanned flight maneuvers required for experimental repeatability.

The primary flight computer tasks include sensor processing, component health and status monitoring, telemetry data processing and servo-control. Sensor processing includes signal filtering, smoothing and redundant sensor data comparison. Health and status will be provided for all elements of the flight control system. Logic should be developed respond to flight critical component failures located on-board the MURV. Data management for the telemetry uplink and downlink will be provided. Finally, the controller-actuator interface, not yet defined, will provide closed loop servo control.

The backup flight control computer performs functions similar to the primary. The backup will not process system status data and will only have telemetry uplink processing requirements. The backup controller should provide both open loop control and preplanned flight modes for independent operation. Possible backup flight modes include:

1. Control recovery
2. Orbit
3. Straight and level
4. Turn
5. Loss of signal maneuver

The first four flight modes were employed in NASA's HiMAT backup flight control system [55]. The loss of signal maneuver is recommended as a contingency. The maneuver would consist of a 180 degree turn with an increase altitude command.

This development formed the basis for the conceptual design of the flight control system. Supermaneuverability flight modes were used to identify some preliminary requirements for sensors and and control surfaces. This ensured that the vehicle configuration, data acquisition system and remote cockpit designs will be able to support the implementation of a flight control system suitable for maneuverability experiments. Where possible data, was presented to provide some level of expectation for future design efforts. The selection of Design #2 was based on the the design objectives. Detailed design efforts are required to develop the flight control system. These tasks are identified in Volume Two, Section 6.4.

V. Launch/Recovery System

5.1 Problem Definition

5.1.1 Introduction Integral to the design of the MURV is the capability for the vehicle to be safely launched and recovered for further flights. This capability requires the development of a launch/recovery system that will allow the MURV to be successfully launched and recovered by its operators.

5.1.2 Scenario At this point in the development of the MURV the most promising location for operations is the Jefferson Proving Grounds, located approximately 120 miles southwest of Wright-Patterson AFB (see Appendix N). This location offers a paved (asphalt) runway, 40 feet wide and 600 feet long. Additionally, it is surrounded by an area of approximately 70 square miles for the MURV to operate within. Because a selection of an aerodynamic configuration had not been selected at the time of the analysis for this section, any aerodynamic performance data required was either averaged between the three candidate configurations or representative data was used. We felt that based on the focus of the initial launch/recovery system selection process that this reasoning was justified.

5.1.3 Scope The initial development focused on deciding on the best overall launch/recovery method. This decision was based on the top level objectives of the launch/recovery system and the analysis techniques used were cursory. A systematic systems approach similar to that described in Section 2.1 was used to evaluate and choose the most promising method during the first cut. After this decision was made additional development using the selected aerodynamic configuration was accomplished. The detailed discussions of this work can be found in Section 5.5.

5.1.4 Needs The two most important needs for the launch/recovery system are that it must provide a capability for a safe and stable launch and recovery of the air vehicle. "Included in its ability for launch and recovery in a safe and stable manner is that the landing gear must absorb the energy of landing, provide speed control, directional control during acceleration, deceleration and taxi, and be statically and dynamically stable"[76:1]. The next two needs are inherrent to all of the MURV's subsystem designs. They entail the need for the equipment to be light weight and compact. Air vehicle designers are continually attempting to squeeze the last ounce of weight from separate components without affecting the safety, and reliability of the entire vehicle. Next, it must be low cost. As with the need to be low weight and low volume, landing gears are typically considered as extra baggage on an aircraft. However, great care has to be made to ensure that an adequate system is designed up front so that maintenance and repair costs do not constitute a problem. Finally, the launch/recovery system must not be overly complex. A simpler landing gear, one without extensive kinematics (retraction), or with a small number of components would be favored over a more complex design with a large number of components. This last need fits roughly into the system's reliability and maintainability.

5.1.5 Constraints The first constraint placed on the launch/recovery system involved the volume and weight of the aircraft. Because the process of aircraft design involves many other subsystems, great attention has to be given to ensuring that the weight and volume taken up by the landing gear does not impinge on another subsystem. For the first iteration of the landing gear design this constraint was still very tentative, but as the design of the MURV becomes more defined greater attention to this constraint will take place. The next constraint is a byproduct of the first objective of the overall MURV system relating the versatility of the MURV for different types of tests it can accomplish (see Section 2.2.3) and the primary design driver to maximise the testing of a unique type of experiment involving supermanueverability (see Section 3.1). It requires that the landing gear must allow the MURV to have the capabilities for clean aerodynamics. By clean aerodynamics it is meant that the landing gear has either retracted into the aircraft, been jettisoned after launch, or has in some fashion not exposed itself to the airstream around the aircraft. The next constraint involves the operational complexity needed with the launch and recovery operation.

Because the MURV is to be operated by AFIT personnel and students, the flight operations can not exceed the capabilities of its operators or require extensive outside effort and coordination. The final constraint is the use of the Jefferson Proving Ground as the testing area for the MURV. This constraint specifies an asphalt runway 40 feet wide and 600 feet long for those launch/recovery systems that do require a landing strip.

5.1.6 Other considerations There are several other considerations to be used in the selection process. The first of these involves the initial testing of the MURV. The initial flight tests of the MURV will include extensive handling qualities testing and pilot proficiency tests. For a launch/recovery system that requires use of a landing strip a considerable amount of testing to include high speed taxi tests or touch and gos is expected. This would tend to favor a design that exhibits a very stable platform during acceleration and deceleration. For a design that either quickly accelerates the aircraft to flying speed or by some other means rapidly transitions the MURV to flying attitude, attention must be given to the safe controllability by the operator and the possibility for loss of control without enough time to recover (e.g. a catapult launch that imparts a downward moment that cannot be corrected before impact). Another consideration is that of foreign object damage (FOD). Initial decisions for the design of the MURV stipulated the use of a turbojet engine with an inlet located at the bottom of the fuselage. During the takeoff and landing phases for the MURV the inlet acts as a very efficient vacuum and any small items on the runway have a potential of being ingested into the engine causing damage to the engine's internal components. Therefore, some way to either keep the MURV's inlet at a large enough standoff distance from the runway surface or some other method to protect the engine from FOD must be included in the design.

5.1.7 Summary The purpose of the first four sections of this Chapter is to determine the best launch/recovery system method for the MURV. Various concepts will be identified and analysed based on the systematic systems approach using the needs, constraints, and the other considerations identified above as they apply to the initial iteration of the design process. The culmination of this initial iteration will be one or possibly two concepts which will be fully assessed and additional detailed design work accomplished.

5.2 Value System Design

An objective hierarchy method was used to develop the objectives of the launch/recovery system for the MURV. The very highest objective is to design the best launch/recovery system for the MURV. The objectives that branch out from this objective are derived from the needs identified previously. They are:

1. Provide a safe and stable launch of the MURV;
2. Provide a safe and stable recovery of the MURV;
3. Minimize weight;
4. Minimize volume;
5. Minimize complexity;
6. Minimize cost;
7. Maximize modularity.

These seven objectives make up the first level of the objective hierarchy tree as shown in Figure 5.1.

5.3 System Synthesis

A quasi-morphological approach was used to generate alternative solutions to the design problem. In this approach, research into various launch and recovery methods that could apply to the MURV was carried out. After completion of this research a brainstorming session was used to create a wide variety of separate launch system design methods and recovery system design

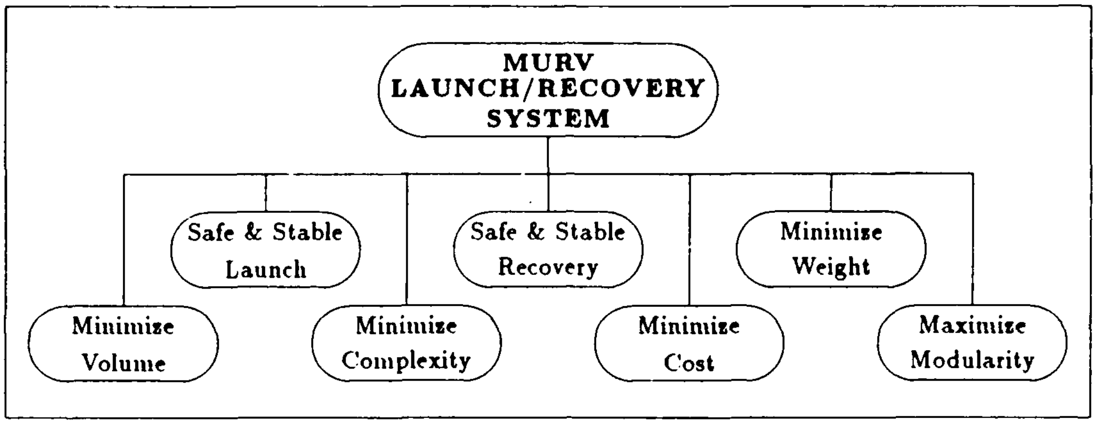


Figure 5.1. Launch/Recovery System Objective Hierarchy Tree

methods. Then by applying a quasi-morphological approach various launch/recovery combinations were formed. This approach entailed the matching of compatible launch and recovery systems to form an integrated launch/recovery system. The following two sections illustrate the various launch and recovery design methods that were considered.

5.3.1 Launch System Descriptions

5.3.1.1 Conventional Fixed Undercarriage (L1) The tricycle style (two main wheels located behind the aircraft center of gravity with a single front nose wheel) was the only type of wheeled undercarriage considered. A tail-dragger configuration (two main wheels located ahead of the center of gravity and a single tailwheel) has inherent stability problems that have led modern day aircraft designers to all but abandon this configuration [20:7]. Because of the need for the operator to be able to control the vehicle as it accelerates during the takeoff, and decelerates during the landing, a means for directional control must be included. A steerable nose gear is the most widely used method and therefore it is a part of this type of launch method. A major variation to this design is the use of brakes on the main wheels. A typical conventional fixed tricycle undercarriage used in the Israeli, Mazlat Scout 800 is shown in Figure 5.2.

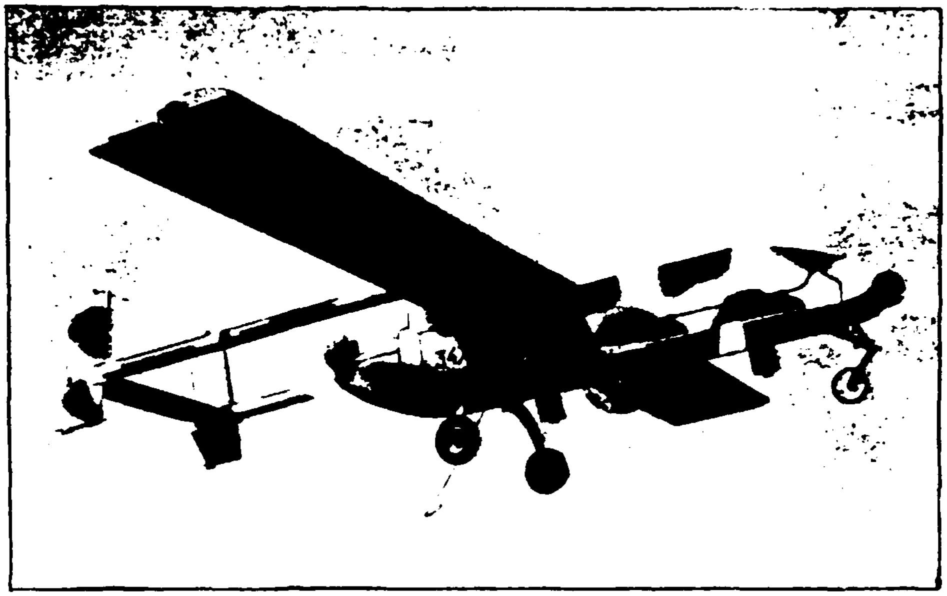


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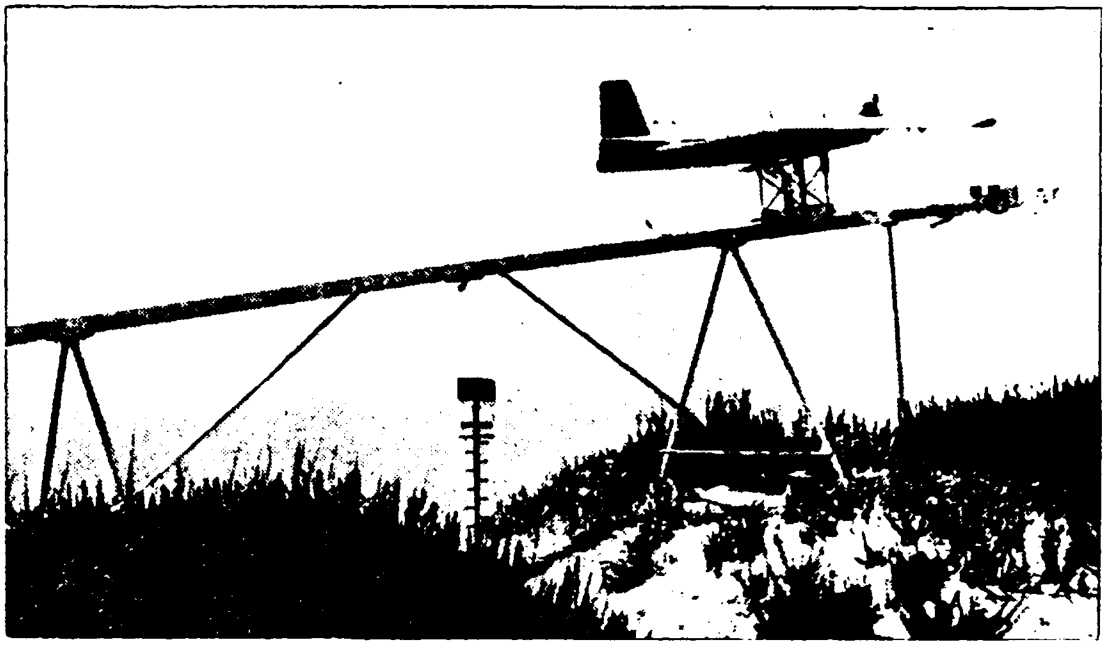
Figure 5.2. Mazlat Scout 800

5.3.1.2 Retractable Undercarriage (L2) As with the conventional fixed undercarriage only the tricycle configuration was considered. The major difference between this type of undercarriage and the fixed undercarriage is that a retraction system will tuck the external wheels and struts into either the fuselage or wing and therefore provide a smooth surface external to the aircraft. A major part of the modularity concept is the use of a relatively standard fuselage design that can support various types of wing/tail/canard combinations (see Section 7.2). By having the wheels retract into the wing would severely restrict the placement of the undercarriage and/or wing. For example, if a modification of the wing mounting of the MURV from low to high was desired, the landing gear would have to be extensively redesigned. Also, a change in the location of the wing either fore or aft could violate placement of the rear landing gear struts with respect to the center of gravity of the MURV. Therefore, for the design of MURV, retraction into the wing was not considered. Because this launch method has the same requirement for directional control as L1, steering is incorporated into the nose gear. As with the the fixed undercarriage, variations for with or without main wheel brakes were included.

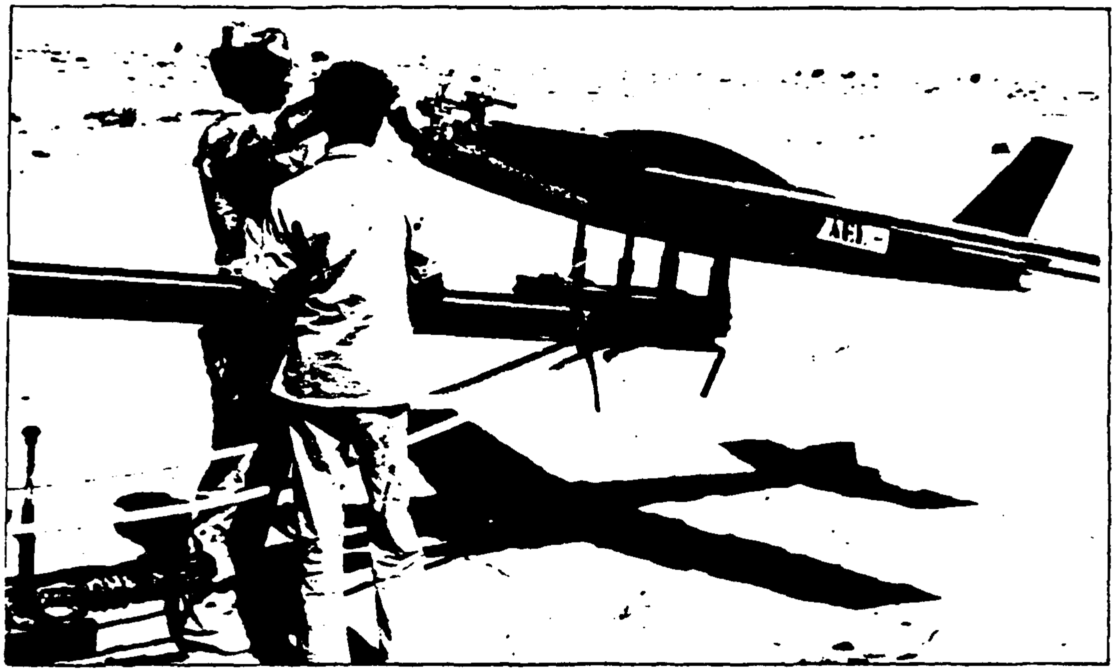
5.3.1.3 Rail Launch (no catapult) (L3) This launch method would entail mounting the MURV to a dolly or similar device which would support the aircraft on a rail while it is accelerating to flight speed. Once the MURV had achieved takeoff speed it would be released from its support and climb away from the rail. This method would require a relatively long rail (well over 100 feet) to allow the MURV to fully accelerate to flight speed and maintain a climb after release of the support. Figure 5.3 shows a remotely powered vehicle (RPV) utilizing this method of launching.



[94:363]

Figure 5.3. Tamnar EDO on Rail Launcher

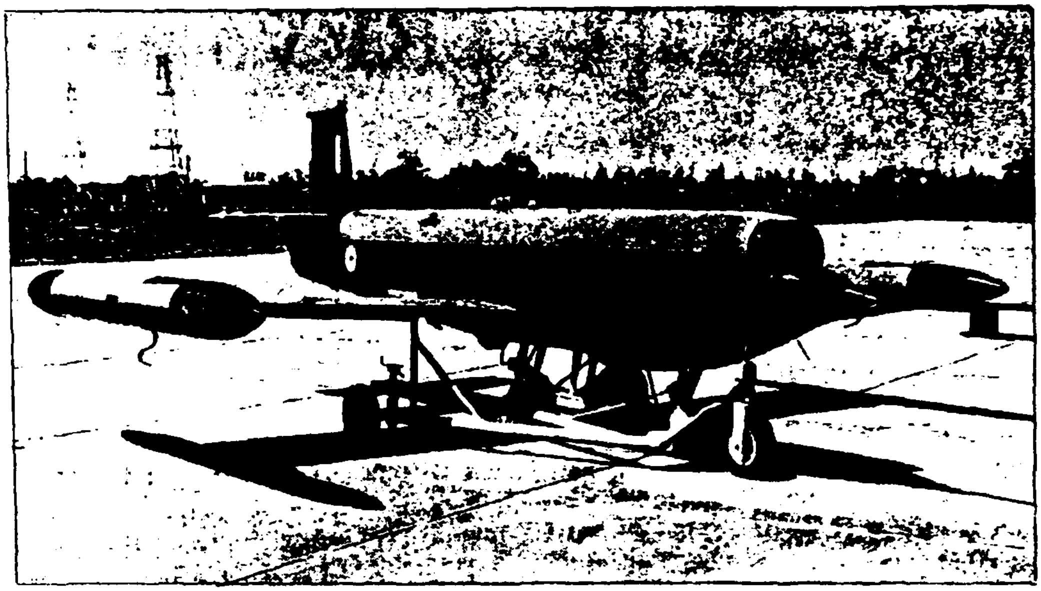
5.3.1.4 Rail Launch (catapult) (L4) This type of launch is essentially the same as the previous design with the exception of a much shorter rail and the inclusion of a catapulting device that would accelerate the MURV much faster than the capabilities of the MURV's internal engine(s). Because of the short transition time for the acceleration of the MURV from rest to flying speed and the possibility of an abnormal attitude immediately after the vehicle leaves the rail, this design probably would impact the design of the flight control system and/or the sizing of the control surfaces as well as the abilities of the operator. Many of the current RPVs such as the British AEL 4700 Snipe Mk III (see Figure 5.4) use this type of launch system.



[95:799]

Figure 5.4. AEL 4700 Snipe Mk III on Catapult Launcher

5.3.1.5 Dolly Undercarriage (L5) This launch design would include a typical tricycle wheel arrangement as discussed before, built into an autonomous dolly detached from the aircraft. The MURV would be supported by the dolly during ground operations and during its acceleration down the runway. Upon the point of reaching liftoff speed the dolly would be released and the aircraft would climb away. The steering of the nose wheel could be accomplished by one of two methods. One method would use a steering mechanism built into the dolly itself and controlled by the operator. The other method would interlink the aircraft rudder controls mechanically (or electronically) with the nose wheel steering on the dolly so the two would work in sync. The use of a dolly would allow the rear main wheels to be separated by a larger distance than either the fixed or retractable gear enhancing its stability on the ground, and during acceleration for crosswind conditions. One additional consideration in the design of this type of launch system was the need to include some way to retard the dolly after separation to ensure the dolly does not impact the aircraft at the point of separation, and to bring it back to rest. Shown in figure 5.5 is the takeoff dolly for the Australian, GAF Jindvik Mk 4A.

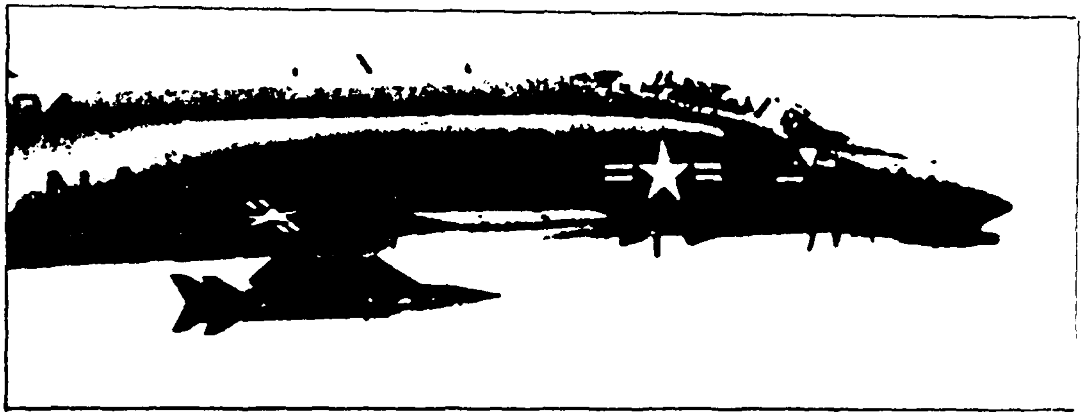


[95:786]

Figure 5.5. GAF Jindvik Mk 4A

5.3.1.6 Rear Wheel Dolly with Nose Wheel Retract (L6) This launch system design incorporates a detachable dolly for the rear main wheels and an onboard front wheel with a retraction device to form a tricycle arrangement. As with the fully retractable launch system steering is incorporated in the nose wheel. Similarities with the entire dolly undercarriage (L5) include the ability to have a large distance between the main wheels and the need to retard the dolly after separation.

5.3.1.7 Air Drop (L7) This technique would require the MURV to be supported by another aircraft that would takeoff and fly to a sufficient altitude with the MURV carried under the fuselage or on a wing pylon and release the MURV to begin flying on its own (similar to some of the X-series aircraft and the Air Launched Cruise Missile (ALCM)). Major considerations of this type of design are the need to provide some means of starting the engine while supported by the carrier aircraft and the availability of this carrier aircraft for AFIT use. Figure 5.6 shows a USN AQM-37 Variant target drone that uses this type of launch system.



[95:807]

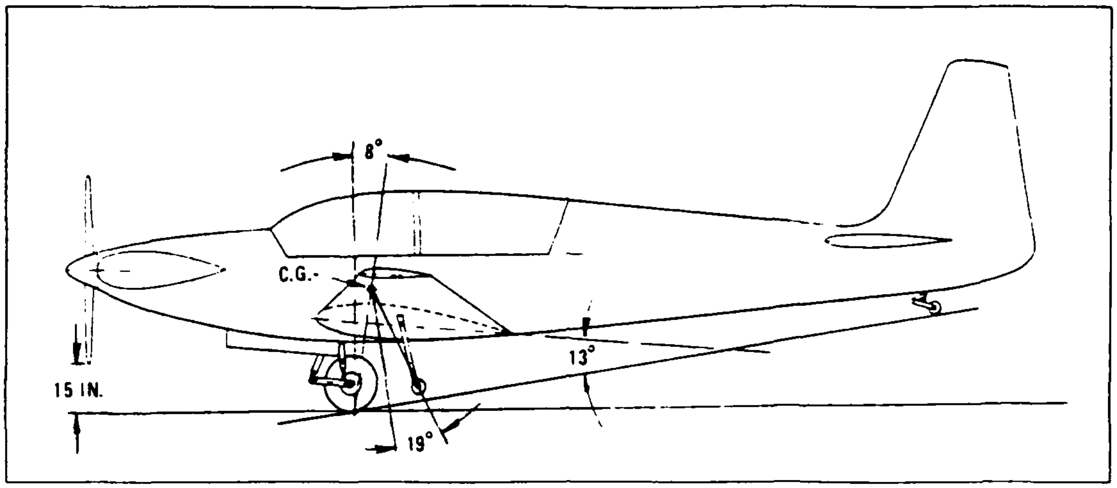
Figure 5.6. AQM-37 Variant Target carried by a US Navy QF-4B Phantom

5.3.1.8 Centerline Wheel(s) with Tip Wheels (L8) This launch system design incorporates wheel(s) located on the bottom centerline of the fuselage with wheels mounted off the wingtips (if necessary) to balance the aircraft. This type of gear is commonly used on high performance gliders and some motorised gliders as shown in Figure 5.7.

5.3.2 Recovery System Descriptions

5.3.2.1 Conventional Fixed Undercarriage (R1) This design is essentialy the same as the launch system (L1) with the exception of the possibility of including a drogue chute to the non brake version to aid in stopping the MURV after touchdown.

5.3.2.2 Retractable Undercarriage (R2) This design is essentialy the same as the launch system (L2) with the exception of the possibility of including a drogue chute to the non



[76:17]

Figure 5.7. Sportavia RF-5B Motorglider with Center Wheel

brake version to aid in stopping the MURV after touchdown.

5.3.2.3 Net Recovery (R3) This design would involve the use of a recovery net on the ground. The operator of the MURV would be required to fly the aircraft into the net which would bring the aircraft to rest (see Figure 5.8). The net to capture the aircraft would have to be made large enough to ensure that the operator could in deed safely fly the MURV into the net.and strong enough to recover the vehicle without causing damage.

5.3.2.4 Skids (R4) This landing method would use rear mounted skids to absorb the weight of the aircraft upon landing and a forward wheel to facilitate steering and provide a stable platform on the ground. It is essentially the tricycle arrangement (R2) above, with skids replacing the rear main wheels. Because the skids will act as braking surfaces upon contact with the ground no other braking device is needed. For this design the skids would be pretracted in the aircraft prior to landing and then deployed with a downward only, acting (no need to retract) device. Pretracted means that because the mechanism to deploy the landing gear only acts in the direction to deploy, the rear skids are stored in the "up" position. This allows the use of a preloaded spring device

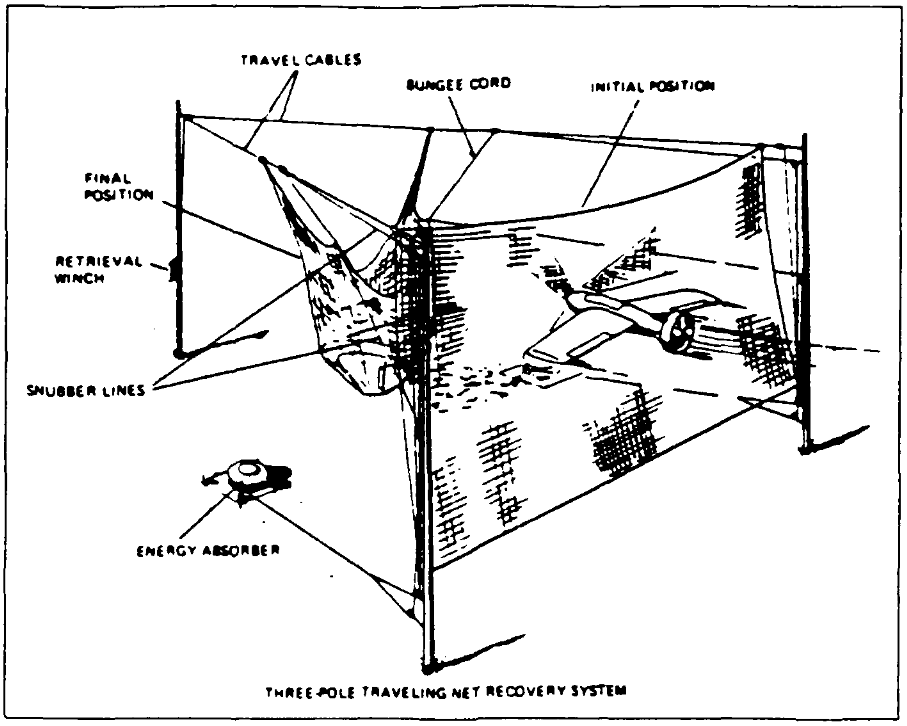
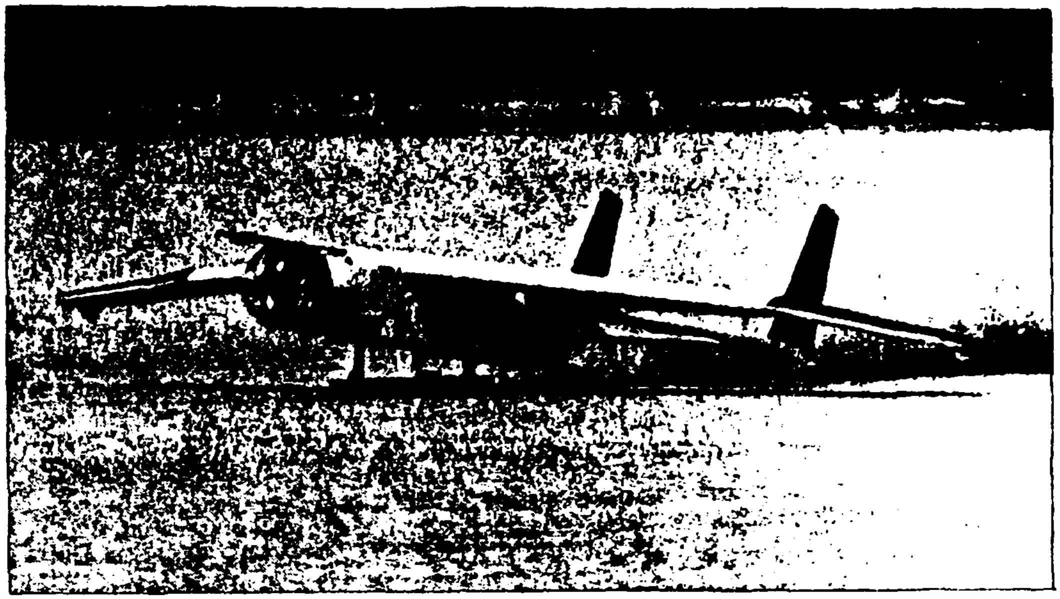


Figure 5.8. Aquilla Net Recovery

to deploy the skids instead of a mechanical device requiring power inputs. The nose wheel would employ either a deployment only design or deployment/retraction design depending on whether the nose wheel is required during takeoff or not. Figure 5.9 shows the SkyEye R4E-40 making a skid landing.

5.3.2.5 Mid Air Recovery (R5) This design would use a parachute to descend the MURV at a slow enough velocity that a helicopter could retrieve it by snaring the parachute. This type of recovery system has been successfully used by the ALCM for flight test recoveries.

5.3.2.6 Parachute Recovery (R6) This recovery method would use a large parachute to descend the MURV to the ground. A way to control the flight path of the parachute would probably also be required to ensure that the landing point of the aircraft could be partially controlled. Additionally some device such as an airbag on the bottom of the fuselage is needed to absorb part of the landing shock.



[95:813]

Figure 5.9. LSI/DS SkyEye R4E-40

5.3.3 Formulation of Alternatives Now that various launch and recovery methods have been identified, combinations of the two were assembled into potential launch/recovery system alternatives as seen in Table 5.1 and Table 5.2. By using a quasi-morphological method a total of 48 candidate launch/recovery systems were assembled. The next step taken was to screen these candidates based on their respective feasibility.

Table 5.1. Launch/Recovery System Alternatives

LAUNCH			RECOVERY							
	Brakes		R1		R2		R3	R4	R5	R6
Method	Y	N	DC	N/A	DC	N/A				
L1	X			X						
		X	X							
		X		X						
L2	X					X				
		X			X					
		X				X				
L3	X					X				
		X			X					
		X				X				
							X			
								X		
									X	
										X
L4	X					X				
		X			X					
		X				X				
							X			
								X		
									X	
										X
L5	X					X				
		X			X					
		X				X				
							X			
								X		
									X	
										X

Legend

LAUNCH METHODS	RECOVERY METHODS
L1: Conventional Fixed Undercarriage	R1: Conventional Fixed undercarriage
L2: Retractable Undercarriage	R2: Retractable Undercarriage
L3: Rail Launch (no catapult)	R3: Net Recovery
L4: Rail Launch (catapult)	R4: Skids
L5: Dolly Undercarriage	R5: Mid Air Recovery
	R6: Parachute Recovery

Y: Yes N: No DC: Drag Chute N/A: Not Applicable

Table 5.2. Launch/Recovery System Alternatives (cont)

LAUNCH			RECOVERY							
	Brakes		R1		R2		R3	R4	R5	R6
Method	Y	N	DC	N/A	DC	N/A				
L6	X					X				
		X			X					
		X				X				
							X			
								X		
									X	
										X
L7	X					X				
		X			X					
		X				X				
							X			
								X		
									X	
										X
L8	X					X				
		X			X					
		X				X				
							X			
								X		
									X	
										X

Legend

LAUNCH METHODS	RECOVERY METHODS
L6: Rear Dolly Undercarriage	R1: Conventional Fixed undercarriage
L7: Air Drop	R2: Retractable Undercarriage
L8: Centerline Wheel(s)	R3: Net Recovery
	R4: Skids
	R5: Mid Air Recovery
	R6: Parachute Recovery

Y: Yes N: No DC: Drag Chute N/A: Not Applicable

5.3.4 Elimination of Infeasible Designs The elimination of a candidate launch/recovery system was based on assessing each design against the constraints defined in Section 5.1.5. Any design that failed any one of these constraints was discarded. The following sections separately address each of these constraints.

5.3.4.1 Flexibility Screening An obvious elimination was that of all designs using the conventional fixed tricycle gear since it violated the constraint to be able to have clean aerodynamics. In other words, because the undercarriage is always exposed to the airstream there is no capability to perform any tests requiring an aerodynamically clean aircraft.

5.3.4.2 Takeoff Screening One of the most important design drivers for a launch/recovery system is the length of runway or rail required to achieve a takeoff velocity. For the MURV this distance is restricted by the length of the Jefferson Proving Ground (600 feet) for those designs requiring a runway. To make a decision as to whether a method was feasible the takeoff analysis by Leland M. Nicolai[75] was used to calculate the takeoff distance. This calculation was based on the force diagram depicted in Figure 5.10.

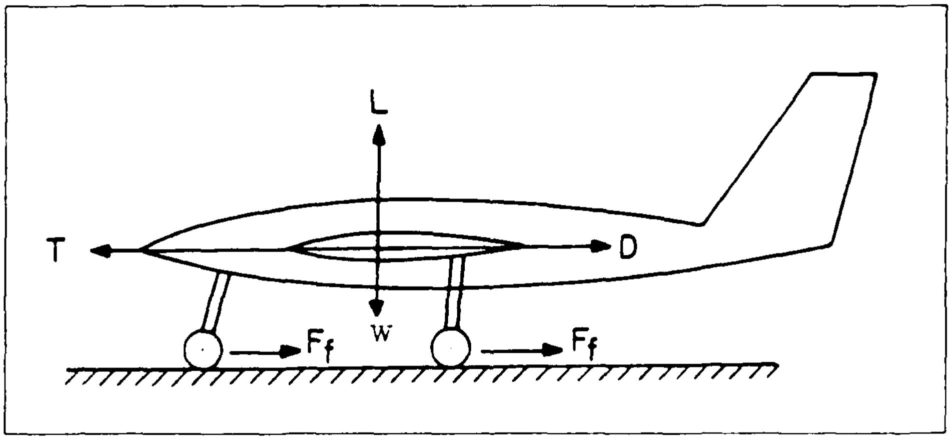


Figure 5.10. Force Diagram During Ground Roll

The derivation of this analysis and the assumptions that were required to do this analysis can be found in Appendix P. Based on this analysis the takeoff distance for designs using a tricycle undercarriage was calculated to be approximately 446 feet. The takeoff distance for the designs using either of the dolly designs assuming a weight of 30 pounds for the dolly and an increase in the drag coefficient C_D of 0.022 was calculated to be approximately 530 feet. Both of these distances do not exceed the runway length at the Jefferson Proving Ground and therefore all of the designs incorporating tricycle undercarriages or a dolly for takeoff pass the feasibility test for the takeoff distance.

The decision to eliminate both of the rail launch designs was based on an extension of the analysis in Appendix P. First, for the launch method using a rail without any catapulting device, the design, fabrication, and implementation of a rail with a length of 446 feet (approximately the same as that for a wheeled undercarriage) was deemed to be extremely impractical. For the launch method using a rail with a catapulting device the same analysis was modified by the following:

1. The acceleration term would have an additional term T' for the thrust of the catapult.
2. The force diagram would be modified to include the rail being elevated at an angle of approximately 10° (see Figure 5.11). The elevation is required to ensure the MURV will be able to safely transition to free-flying flight after the catapulting phase of the launch.

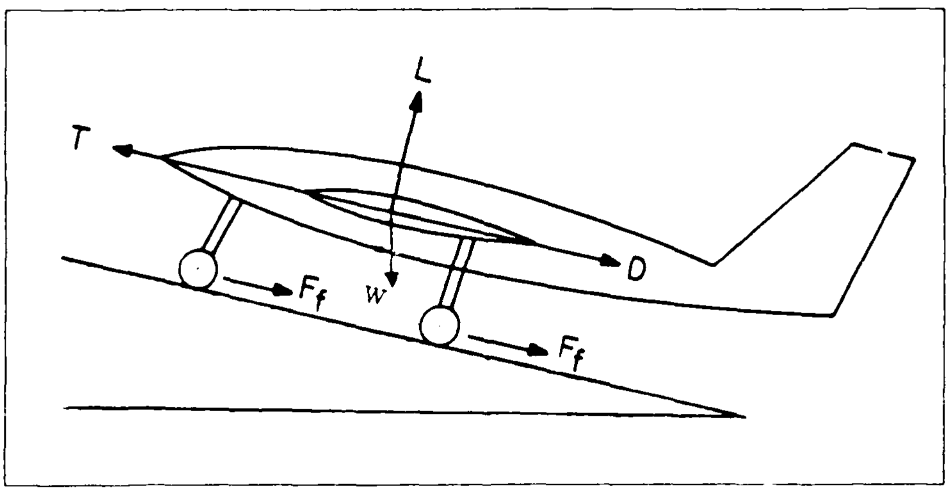


Figure 5.11. Force Diagram Using an Elevated Rail

To evaluate the feasibility of a catapult launch design a maximum allowable longitudinal acceleration of 3g was selected [85:74]. Solving for the length of rail required resulted in a distance of 61.5 feet and an additional thrust required of approximately 580 pounds. Because of the length of the rail required and the complexities involved with the construction of a rail and catapult with 580 pounds of thrust, this method was discarded. Though it would be possible to decrease the length of the rail it would require an additional increase in the thrust of the catapult. This increase in thrust would severely impact the structural design of the MURV. An additional consideration for the elimination was the high probability of fatigue induced failures (due to launches) for the structure of the vehicle. To be able to ensure the structural integrity a considerable amount of analysis or a large increase in the structural safety factors (thereby increasing the structural weight) would be required.

5.3.4.3 Landing Distance Screening Another major design driver in the design of a landing gear is the landing distance required by the aircraft. As with the takeoff analysis for the MURV this distance is once again restricted by the use of the Jefferson Proving Ground. One of the significant differences between many of the remaining candidates is the inclusion of brakes or a drogue chute. To make a decision as to whether any these different methods were feasible the landing analysis by Nicolai M. Leland[75] was followed. Investigating the use of brakes first, the landing distance was defined as the horizontal distance required for an aircraft to settle to the runway (after crossing the runway threshold), apply brakes, if available, and come to a complete stop. Figure 5.12 depicts the partitioning of the landing distance analysis. LD_A is the distance for the approach, LD_{FR} is the distance covered after touchdown, but prior to application of the brakes, and LD_B is the braking distance to bring the MURV to rest.

Many assumptions were required to perform the analysis. These assumptions and the derivation of the analysis can be found in Appendix P. Using this analysis, the landing distance for the MURV for the worst case scenario of having to land with a full fuel load, was calculated to be ap-

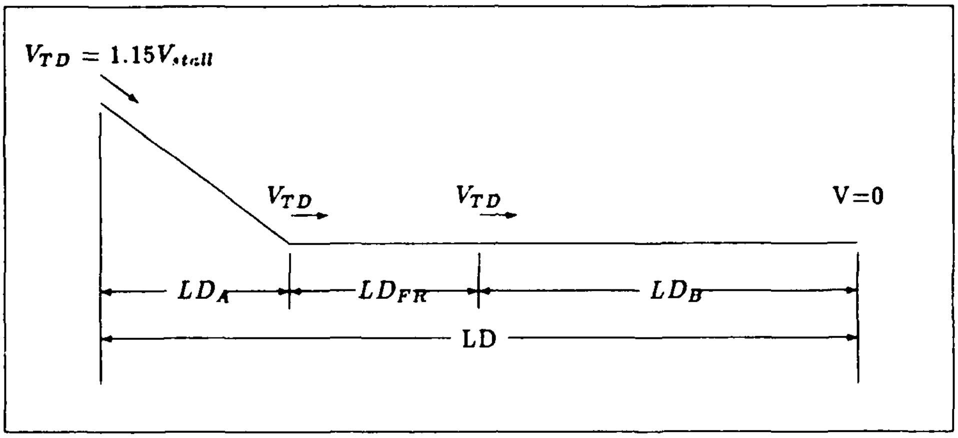


Figure 5.12. Illustration for Landing Analysis

proximately 574 feet. This distance is slightly less than the runway length at the Jefferson Proving Ground and therefore is a feasible design. For the designs utilizing the skid recovery system the landing distance will be significantly shorter than that of a design incorporating brakes since the coefficient of friction would be at least 0.5 at the point of touch-down and increase to a maximum of 0.9 just prior to coming to rest[56]. Therefore, the landing distance for skids will be at worst less then that of braked wheel version.

Next, the situation where there are no brakes and no other retarding forces (i.e. drogue chute) was calculated to be 2637.3 feet. Even if the wing loading is reduced to 8 (approximate zero fuel landing condition) the landing distance is still 1552.4 feet. These landing distances are considerably longer than the length of runway available and therefore all launch/recovery system designs incorporating this type of landing method were eliminated.

The final landing condition examined was the variation of the analysis in Appendix P for the use of a drogue chute to decelerate the MURV. Now, in addition to the original drag due to the aerodynamics of the aircraft there is drag due to the drogue chute. Based on this analysis a launch/recovery system having no brakes but a 6 foot diameter drogue chute would require approximately 577 feet to land and come to a complete stop. Therefore the use of a drogue chute as the retarding device cannot be discarded.

5.3.4.4 Weight Screening Even though the exact values for the allowable weight for the launch/recovery system are still very tentative, an upper limit of 6 % [20:1] of the design gross vehicle weight was selected. For the MURV configurations that were used in this analysis that worked out to a maximum allowable weight of 12.6 pounds. Both the parachute recovery design and the brake configured designs had a high potential for being discarded as being infeasible. The weight of the parachute canopy was calculated by multiplying the weight of material, $\frac{1.4oz}{yd^2}$ for nylon [16:45], by the surface area of a hemisphere. The expression for the canopy weight took on the following form:

$$W_c = 1.53 \times 10^{-2} D^2$$

where W_c is the weight of the parachute canopy and D is the maximum diameter of the canopy. To estimate the entire parachute weight (including tape reinforcements, drogue lines and deployment equipment) a multiplcative factor of two was considered conservative. Therefore, the entire weight of the parachute recovery system (W_p) was estimated using

$$W_p = 3.06 \times 10^{-2} D^2$$

Using this estimation and the maximum allowable weight of 12.6 pounds a diameter of 20.4 feet was calculated. Next, an estimate of the descent velocity was needed to verify that it would not exceed a reasonable impact velocity of 10 $\frac{ft}{s}$[76:125-126]. With a 20.4 foot diameter parachute the terminal velocity was calculated by the following formula [16:7].

$$Z = \frac{1}{D}\sqrt{\frac{8W}{\pi \rho C_D}}$$

where W is the weight of the MURV (approximately 200 lbs., worst case), D is the diameter of the parachute and C_D is the parachute's drag coefficient (approximately 1.4), and ρ is the sea level air density. The result was a terminal velocity of 19.2 $\frac{ft}{s}$ which exceeds the maximum allowable terminal velocity and therefore the parachute recovery method was discarded.

Next, based on data for light aircraft [76:82-84] the weight of the brake assembly was calculated using the following estimation technique. First, the kinetic energy absorption requirement for each main wheel brake asssembly was derived using the following formula:

$K.E. = 0.0443\frac{WV_{stall}^2}{N}$ where

$K.E.$ = kinetic energy per wheel ($ft \times lbs$)

W = design landing weight (lbs)

V_{stall} = stall speed in knots.

N = number of main wheels.

Using previous data from the landing analysis (Appendix P) the K.E. absorption required for the MURV is 10971($ft \times lbs$). Using this value and the available brake assemblies [76:83-84] the lightest weight brake assembly available would weigh 5.44 lbs per wheel giving a total weight of 10.88 lbs just for the wheel and brake assemblies for the two rear main wheels. This obviously would not leave any weight for the retraction mechanisms and forward nose wheel assembly, therefore any launch/recovery system requiring brakes was discarded.

5.3.5 Operational Complexity Screening For the MURV to have a feasible launch/recovery system it must satisfy the constraint of not exceeding the capabilities of the personnel to operate the system. The net recovery method (R5) fails this constraint. At the approach speeds that the current configurations are estimated to be flying at (approximately 107 ft/s), the ability of the pilot to control the aircraft flight path into a net would be taxed too excessively. Additionally, the forces imparted on the vehicle during capture would require a considerable amount of analysis and testing to ensure that the vehicle's structure could withstand them. Similarly, to enable the structure to carry relatively high longitudinal loads would impact the weight of the structural members required. At the time of this study exact figures on the total structural weight are not available, however, the MURV will be hard pressed to meet the requirements of some of the missions envisioned based on early estimates of the overall vehicle weight, and a highly probable increase to the structural weight for the net recovery system was considered as justification to eliminate this design. The mid air recovery system fails the same constraint as above because of the design requirement that includes a helicopter to capture the MURV after parachute deployment. The need for a helicopter exceeds the capabilities of AFIT and requires too large of an effort from an outside agency and the corresponding coordination associated with this effort. For the exact same reasons the air drop launch method also failed. It is beyond AFIT's capability to include a carrier aircraft in the overall MURV system. Additionally, an attempt to modify a current Air Force asset and have it available for AFIT use is beyond the scope envisioned for the MURV. The final design to be discarded was the centerline mounted wheel design. This type of landing gear is extremely difficult to operate for any distance during the ground roll while accelerating or decelerating. It has been widely used in gliders since they have a relatively short takeoff and landing rolls. For an aircraft such as the MURV which will require more than 400 feet for takeoff and landing, it would exceed the capabilities of the operator to control the MURV during the critical operating periods of launch and recovery. An additional shortcoming of this type of launch/recovery system is its inability to handle cross-wind takeoffs and landings.

5.3.6 Summary The initial number of possible alternatives was reduced from an original total of 48 designs to the current number of five feasible designs. The following is a description of these remaining alternatives.

1. A retractable undercarriage without brakes but with a drogue chute;
2. A three wheel dolly undercarriage with a pretracted tricycle landing gear without brakes but with a drogue chute;
3. A three wheel dolly undercarriage with a pretracted set of rear skids and front wheel;

4. A rear two wheel dolly undercarrige with a retractable front wheel and pretracted rear wheels without brakes but with a drogue chute;
5. A rear two wheel dolly undercarriage with a retractable front wheel and a pretracted set of rear skids.

5.4 Final Design Selection

5.4.1 Introduction To be able to evaluate the five feasible configurations, models had to be developed to differentiate between the designs. Based on the analysis in system synthesis, the following objectives from the launch/recovery objective hierarchy:

1. Provide a safe and stable launch for the MURV;
2. Provide a safe and stable recovery for the MURV;
3. Maximize modularity;

were identified as not being able to differentiate between designs. Based on the analysis in Sections 5.3.4.2, 5.3.4.3, and 5.3.4.4 and in Appendix P all five of the candidates should have no difficulting operating from the runway at the Jefferson Proving Ground. Also, because of the level of the design which each of the five candidates are at we felt that each could equally be designed to produce a safe and stable launch and recovery. The modularity objective was not considered a player because the skid versions could be easily modified at a later time to accept wheels if necessary to perform STOL testing, or any other kind of testing requiring exposed landing gear. Additionally, the cost objective was not used because cost figures were not readily available and because the five alternatives being considered did not appear to pose any problem with cost. This left the objectives to minimize weight, volume, and complexity to evaluate the final five alternatives.

5.4.2 Weight Model The model for the weight objective was constructed by estimating the weights for separate components and then summing the component weights required by each of the designs. The major components required are listed below with an estimate of their respective weights:

1. Two fully retractable main struts (4.0 lbs)
2. Two pretracted main struts (3.2 lbs)
3. Two main wheel assemblies (2.0 lbs)
4. Two main skids (2.0 lbs)
5. One fully retractable steerable nose strut (3.0 lbs)
6. One pretracted steerable nose strut (2.4 lbs)
7. One nose wheel assembly (1.0 lbs)
8. Drag chute (1.1 lbs)

Table 5.3 summarises the components required by each design and its overall weight

Table 5.3. MURV Launch/Recovery System Weight Estimation

		ALTERNATIVE				
COMPONENT	WEIGHT	1	2	3	4	5
Fully Retractable Main Struts	4.0	X				
Pretracted Main Struts	3.2		X	X	X	X
Fully Retractable Front Strut	3.0	X			X	X
Pretracted Front Strut	2.4		X	X		
Main Wheels	2.0	X	X		X	
Skids	2.0			X		X
Front Wheel	1.0	X	X	X	X	X
Drogue Chute	1.1	X	X		X	
Total Weight		11.1	9.7	8.6	10.3	9.2

Based on historical data a weight of 12.6 pounds (6 % of the MURV TOGW) represents a barely acceptable design whereas a weight of 7.0 pounds (3 % of the MURV TOGW) represents an exceptional design. Using this information and fitting this data to a systems utility function [5:212] the following ratings were given to the five alternatives. Launch/recovery system 1 was rated considerably below average, 2 was rated average, 3 was rated above average, 4 was rated slightly below average and finally, 5 was rated slightly above average.

5.4.3 Volume Model The model for the volume was based on the differences in volumes between the alternatives. These major differences are listed below:

1. The skid design has the lowest volume because the skids would be used to form the outer surface of the MURV.
2. The inclusion of a drogue chute increases the volume required by approximately 236 in^3
3. The use of rear main wheels increases the volume over the skid design by approximately 57 in^3.

Based on these figures the designs using rear main wheels and a drogue chute have to use an additional 292 in^3 of volume over the designs that use the skids. Because the amount of additional volume required was not too excessive (approximately 0.17 ft^3), it was determined that the designs incorporating skids represented a slightly above average design and those alternatives using rear main wheels and a drogue chute represented a slightly below average design.

5.4.4 Complexity Model The model for the complexity objective was divided into two sections based on our identification of two subobjectives under the complexity objective. The first of these subobjectives calls for the minimization of the Ground Support Equipment (GSE) required during the launch and recovery phase of operation for the MURV. The crux of this model was to identify the GSE required for each of the alternatives and assess a score based on the amount and complexity of the required items. Listed below is the GSE required by each design:

Launch/recovery system 1 (none required).

Launch/recovery systems 2-5. A mechanism to lift the MURV back onto its dolly after landing. This is required to move the MURV to the recovery area after landing. We envision this mechanism as being a very simple manual winch or lever operated device.

Based on the above information, alternative 1 was deemed exceptional whereas 2-5 were assessed as being average.

The second subobjective involved the minimization of the complexity of the onboard equipment to increase the reliability of the MURV. Using a similar method as above we identified the following onboard equipment listed below:

1. Two-way acting rear main struts with wheels. This equipment includes the two main load carrying struts, including shock absorbers; the retraction mechanism; and the wheel assemblies, including bearings, hubs and tires. Additionally a door or doors to cover the wheels and struts after retraction would be required to provide the clean aerodynamics.

2. A two-way acting front strut with wheel. The major components in this equipment include: the strut with its shock absorber and retraction mechanism; the wheel bearing, hub, and tire; and a steering device to be used during ground operations including takeoffs and landings. As with the previous equipment a door is required to cover the strut and wheel after retraction.

3. One-way acting rear main struts with wheels. The major difference between this equipment and 1) above is the simplification in the retraction system. Because the gear is pretracted during takeoff a one-way only acting strut is used for deployment of the rear wheels. Additionally, there is a simplification in the gear doors since they only need to open for landing.

4. One-way acting rear main struts with skids. The only difference between this equipment and that of 3) is that because the skids form the outer surface of the MURV no gear doors are required.

5. A one-way acting front strut with wheel. Because this equipment only needs to deploy for landing a simpler retraction device over the one for 2) can be used. Additionally, a simplification to the door cover is realized because of the one-way deployment for landing.

6. Ballistically deployed drogue chute. This equipment should be self contained in its own canister located somewhere towards the rear of the MURV. Design and fabrication of this equipment would probably be handled by an outside contractor with extensive experience in ballistically deployed parachutes.

To quantify this information for the complexity model we judged each of the above based on their qualitative complexity. A scale from one to ten was used where a one represents very simple equipment and a ten represents very complex equipment. The following values were reached for the above equipment.

1) . . . [9]

2) . . . [7]

3) . . . [7]

4) . . . [8]

5) . . . [6]

6) . . . [6]

To evaluate each of the various launch recovery systems against the complexity objective the values for each of the equipments required was totalled for each of the candidate launch/recovery systems. Table 5.4 identifies what equipment is required for each design.

Table 5.4. Launch/Recovery System (LRS) Equipment Required

	Equipment #					
LRS	1	2	3	4	5	6
1	X	X				X
2		X			X	X
3				X	X	
4		X	X			X
5			X	X		

Using Table 5.4 and the values for each of the separate equipments, the following values were calculated for the alternatives.

LRS 1) = 22

LRS 2) = 19

LRS 3) = 14

LRS 4) = 20

LRS 5) = 15

It was decided that a value of 30 represented a barely acceptable amount of complexity and a value of 5 represented an exceptional amount of complexity. As before, by using this information and fitting it to a systems utility function the following ratings were assessed. LRS 1 was rated as below average, 2 was rated as slightly less than average, 3 was rated as above average, 4 was rated as slightly below average, and finally LRS 5 was rated as slightly above average.

5.4.5 Final Launch/Recovery System Evaluation Now that each of the alternatives have been rated against each of the objectives and subobjectives a method was needed to affix numerical values to each rating so as to be able to evaluate the alternatives against each other. We used a method based on Athey's text *Systematic Systems Approach* [5], whereby values between one and nine were assigned based on the information in Table 5.5.

Table 5.6 summarizes the values that were reached for each of the objectives and subobjectives for all five alternatives.

Table 5.5. Assigned Values

RATING	VALUE
Barely acceptable	1
Below average	3
Average	5
Above average	7
Exceptional	9

Table 5.6. Overall Evaluation Matrix

OBJECTIVE	ASSIGNED VALUES				
	LRS 1	LRS 2	LRS 3	LRS 4	LRS 5
Weight	2.5	5	7	4	6
Volume	4	4	6	4	6
Complexity					
- GSE	9	3	3	3	3
- Onboard	4	5.5	7	5	6.5

By inspection alternatives 2, 4 and 5 can be eliminated because alternative 3 either equals or exceeds the value for every corresponding objective and subobjective of each of the other alternatives. This leaves only two alternatives to carry through to the final step in the evaluation process. The first part of this step involves the weighting or importance of the three main objectives. It was determined that the most important objective was to minimize the weight of the launch/recovery system and therefore it was given the highest weighting factor of 0.5. The next important objective was to minimize the volume required so it was given a weighting factor of 0.3. The final objective was allotted the remaining 0.2. Because the last objective, minimization of complexity, had two subobjectives it had to be further broken down within itself. It was determined that the onboard equipment subobjective had greater importance than the GSE subobjective and therefore they were given weighting factors of 0.6 and 0.4 respectively. The final step in the selection process is to multiply the scores for the objectives of each candidate by their respective weighting factors and summing. The candidate which receives the highest score is the best launch/recovery system method. However, before this final step the scores for each alternatives complexity objective had to be calculated by multipying the weighting factors for each subobjective by the corresponding values. This calculation yielded a score of 6 for LRS 1 and a score of 5.4 for LRS 3. What this accomplished was the normalization of scores for each of the major objectives for the final two alternatives. The final results yielded a total score of 3.65 for LRS 1 and 6.38 for LRS 3. Based on these results we decided to drop LRS 1 and to further develop the design of LRS 3.

5.5 Launch/Recovery System Detailed Design

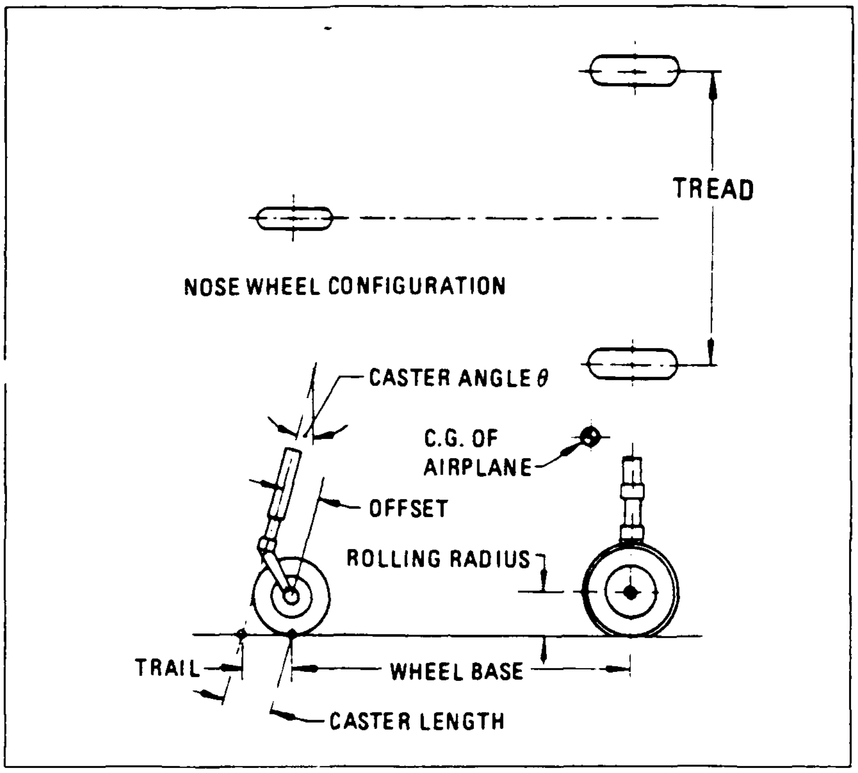
5.5.1 Introduction This section follows a more traditional aircraft design process over the systematic design approach that was used to select the preferred launch/recovery system. The launch/recovery system that we selected can be effectively broken into two major components; one to launch the MURV, and one to recover it, after completion of its mission. The first major component is a three wheeled dolly that supports the MURV during ground operations (fueling, engine start,taxi, *etc.*) and during the acceleration of the ground roll during takeoff. The second major component is the onboard landing equipment. It is comprised of a rear set of skids located behind the engine inlet and a front mounted wheel located just aft of the nose cone bulkhead. For both of these components a tricycle arrangement for the undercarriage was previously selected. The following is a listing of the nomenclature for a tricycle landing gear arrangement[76:6-7] as depicted in Figure 5.13.

Caster Angle - The angle between the centerline of the front wheel spindle and a perpendicular to the ground. (Positive when the lower end of the spindle is forward of the upper end).

Caster Length - The distance measured perpendicular to the spindle axis from the center of the front tire contact area to the spindle axis. Positive to the rear of the spindle axis.

Offset - The distance of the front wheel axis relative to the front wheel spindle axis. Positive to the rear of the spindle axis.

Rolling Radius - The distance from the wheel axle to the ground under dynamic loading.



[76:6]

Figure 5.13. Landing Gear Nomenclature

Wheel Base - The distance measured in a horizontal plane between the front wheel axle and a vertical plane containing the rear wheel axle, with nose wheel on ground in the static attitude.

Tread - The lateral dimension between the centroids of the right and left tire contact areas.

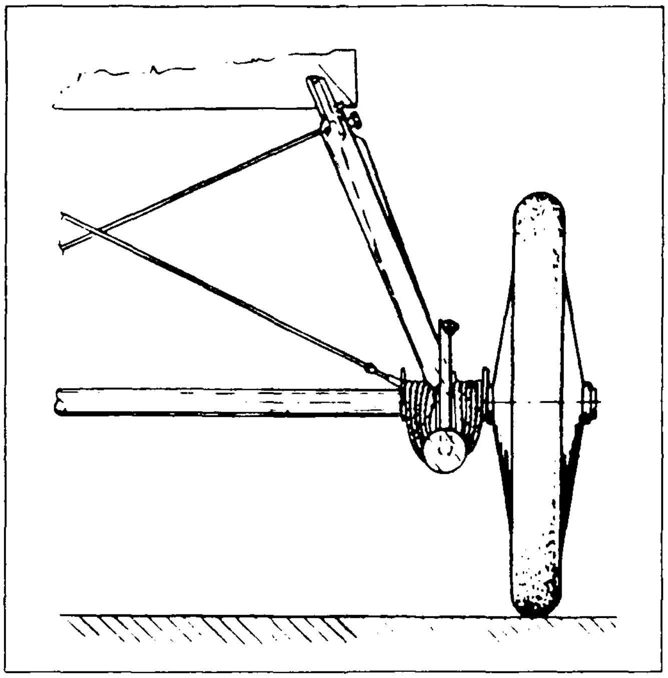
Trail - The distance from the spindle axis projected intersection with the ground and the center of tire contact area.

5.5.2 Launch Equipment As discussed previously, the dolly for the MURV is based on a typical tricycle undercarriage arrangement where the two main wheels are located behind the MURV's center of gravity and a single wheel is located near the front of the MURV. Because the dolly is detached from the MURV itself its impact to the vehicle design is negligible. The actual positioning of the rear main wheels and the front wheel for the dolly is not constrained by the dimensions of the MURV and therefore it can be easily be made both statically and dynamically stable. To better explain this component of the launch system the following sections have been separated into major subcomponents of the three wheeled dolly. They are:

1. The rear main wheels;
2. The front wheel;
3. The dolly frame and cradle;
4. A retarding device;
5. An accelerating device.

5.5.2.1 The Rear Main Wheel Assembly A considerable advantage for using a dolly that separates upon takeoff is the capability of having a relatively large distance (tread) between the two rear main wheels. In the case of the MURV, where wing mounted struts were ruled out, this enables a significant increase in the stability of the MURV during the ground roll for takeoffs in comparison to struts that are mounted to the fuselage. An additional advantage is the capability to have the dolly at a higher height above the ground than a set of comparable struts attached to the MURV. This advantage is significant for the MURV because of the type of engine selected for the baseline. During the takeoff ground roll the inlet to the turbojet will be producing a considerable amount of suction that can pick up debris (FOD) from the runway that can potentially cause damage to the internal components of the engine. We decided that a screen on the cradle (to be discussed later) provided the best solution to the FOD problem and therefore the height at which the dolly had to support the MURV was not constrained by FOD considerations. This is not the case of the onboard equipment discussed in Section 5.5.3. However, some consideration has to be given to ensure an adequate ground clearance during the ground roll. A distance of 12 inches from the bottom of the inlet to the ground will provide adequate ground clearance while not moving the center of gravity of the MURV to high.

Because we currently favor the Jefferson Proving Grounds as the operating site for the MURV, a hard surface runway will be used for takeoffs. The steering will be accomplished using the nose wheel, therefore, we ruled out the need for differential braking of the rear wheels. This tremendously simplifies the design of the suspension and the wheel assemblies for the rear main wheels. For the suspension, either a straight axle with a bungee cord for suspension as shown in Figure 5.14 or a cantilever spring gear as shown in Figure 5.15 could be used.

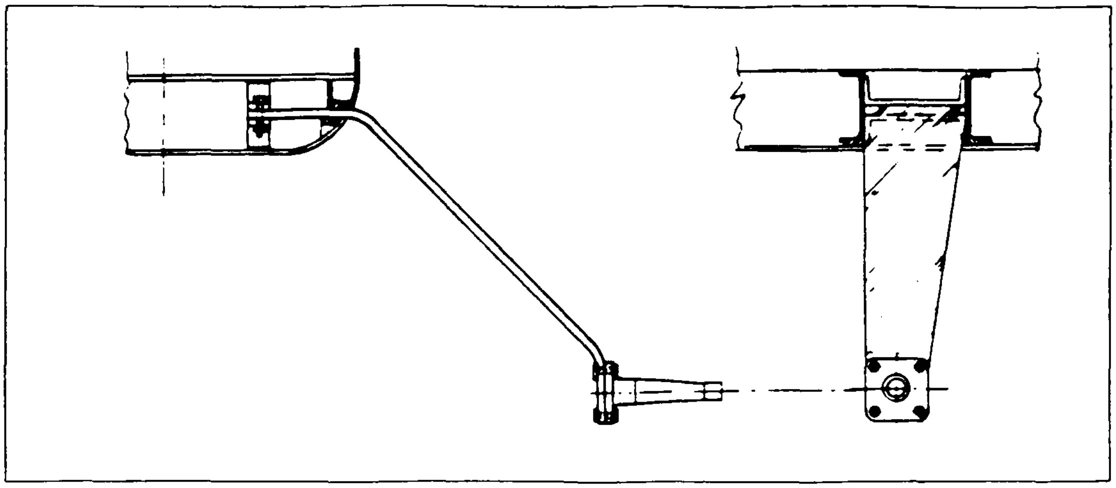


[76:137]

Figure 5.14. Landing Gear with Bungee Cords

Because of the availability of an asphalt runway, something as simple as the wheels and bearings from an all-terrain bicycle could be used for the main wheels. A final consideration for the wheels is the possibility of using wheels that do not require an inner tube, thereby, eliminating the possibility of a flat tire during the acceleration ground roll.

5.5.2.2 The Front Nose Wheel Assembly The basic components of the nose gear are similar to those of the main gear with the exception of the need for a steering mechanism to allow

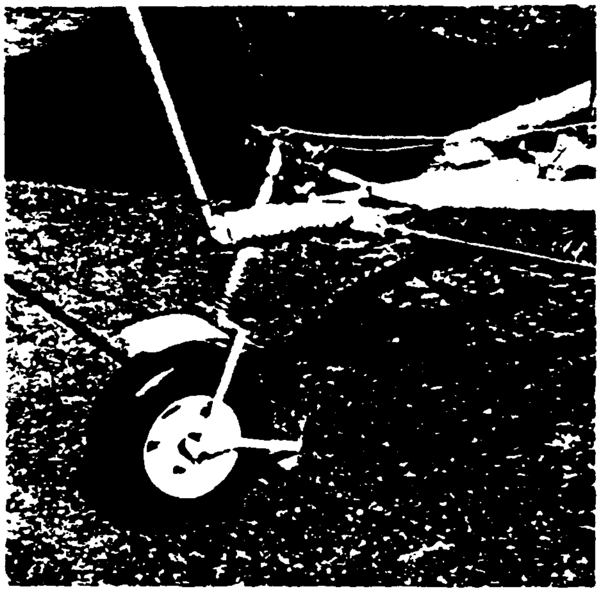


[76:150]

Figure 5.15. Cantilever Spring Gear - Fuslage Attachment

the operator to keep the MURV centered on the runway during the acceleration roll. One variation that was left in this design from the previous design selection process section was the method of linking the steering of the dolly nose wheel with the rudder on the MURV. This linkage is critical for the takeoff because the operator uses inputs to the dolly steering for low speed steering but needs to smoothly transition to inputs to the rudder as airspeed increases. This is due to the fact that the rudder has very little control authority at low airspeeds whereas the nose wheel has its highest control authority at these same low airspeeds. As the airspeed increases this control authority reverses between the two. Therefore it is essential that the operator should not have to switch from controlling one to the other during the takeoff itself. The first variation used a separate servo on the dolly, working in sync, but physically separated from the rudder controls onboard the MURV. The other variation would physically connect the control inputs to the rudder with inputs to the dolly's nose wheel. Because of the added complexity to the MURV/dolly interface and the additional increase of output power that would be required of the onboard servo, we decided that the first variation was the better design. Additional development work will have to be done to ensure that the rudder inputs match the inputs on the dolly. Because the front wheel steering will receive control inputs separate from those to the MURV's on board receiver, part of the nose wheel design will include a receiver and a servo to control the steering of the nose wheel.

As far as for the physical arrangement of the nose gear, the design can be kept relatively simple. Once again due to the use of an asphalt runway many choices are available for the nose gear. The most common type of nose gear used by ultralight and light aircraft is a simple telescopic gear as seen in Figure 5.16. This telescopic strut can either use a helical compressive spring or an oleo-pneumatic (standard automotive) shock absorber. In either case there are many available manufacturing sources and they both provide ample shock absorbtion characteristics for the MURV during takeoff. Because the nose wheel of the dolly will not be experiencing any loads associated with landing the caster angle, length and offset are not pertinent to the design. In fact the nose gear can be mounted vertically to simplify the design.



[76:150]

Figure 5.16. Telescopic Nose Wheel Assembly

The same reasoning that was used for the main wheel bearings and assemblies applies to the nose wheel assembly as well. However, a better set of bearings over those used by the main wheels is needed because of the steering required for the nose wheel. However, many commercially available bearings could be easily incorporated into the design of the nose wheel assembly.

The final consideration for the nose gear design applies to the main gear as well. "The normal nose wheel load is in the order of 10 to 20% of the aircraft weight. Too much nose wheel load may require high elevator down load to rotate the aircraft during the takeoff. A too light nose wheel load will make steering inadequate. The best compromise is to have approximately 15% of the weight of the aircraft on the nose wheel . . ."[76:10]. This is more applicable to the design of the on board landing gear but it is useful to make some points for the dolly design. Since the MURV will be releasing the dolly once it has reached sufficient airspeed to climb away from the dolly it is not as critical a design parameter. Therefore due to other design constraints to be discussed later we decided that the nose gear should bear a larger amount of the load approaching as much as a third of the total load.

5.5.2.3 The Dolly Frame and Cradle The cradle of the dolly must be designed so that it matches the contour of the bottom surface of the MURV's fuselage. It can not rise up high on the side of the fuselage because of the location of the wings when they are mounted low. Also physical clearance for the inlet must be provided and the airflow into the inlet must not be restricted. Based on the type of engine selected for the baseline some means to protect the engine from FOD must be incorporated into the dolly design. The engine that we have selected could ingest a limited amount of sand particles without any substantial loss of thrust[3]. However, if FOD does occur, internal damage may require an engine overhaul. One of the most favorable designs is to use a screen cover that can protect the entrance to the inlet while not restricting the airflow into the inlet. Many screen materials such as those used for window and door screens could be suitably used for this purpose. By providing an open area within the cradle and using this type of material, the probability of FOD could be reduced to insignificance.

The support of the MURV would be provided by cross braces that conform to the MURV's fuselage. These crossbraces would be connected to a frame that also connects to the front nose gear and rear main gear assemblies. This frame must be designed to withstand the maximum takeoff gross weight of the MURV with a conservative 1.5 load safety factor. This frame is envisioned as being nothing more complicated than a truss design using aircraft grade aluminum tubing for its members and gussets for reinforcing wherever necessary.

Another feature that must be included in the cradle design is a means to keep the MURV attached to the dolly until the point where takeoff velocity has been reached and the MURV can climb away from the dolly. It might be possible that friction forces alone might be strong enough to ensure the MURV will not begin to move off the cradle prematurely. If this is not the case either a latch or some other means to hold the MURV down must be included. Additionally, the design for the release of the dolly must ensure that the path of the inlet will not come in contact with any part of the dolly as the MURV lifts off. This should not be a problem because the relative difference in velocities between the dolly and the MURV at liftoff should be small enough that the bottom of the inlet (only 3.55 inches below the bottom of the fuselage) will have ample clearance as the MURV lifts off. However, additional analysis is required to verify the preliminary design.

5.5.2.4 Retarding Device Based on the takeoff analysis that was done in Section 5.3.4.2, the MURV will require approximately 530 feet of the runway at the Jefferson Proving Ground before the release of the dolly. This only leaves 70 feet of runway to bring the dolly back to rest. A slight over run of the runway could be tolerated by the dolly because of its inherent simplicity and strength. An additional 50 feet was allowed for this over run. There were several likely candidate solutions to this problem. The first would be to equip the main wheels of the dolly with brakes that apply a braking force once the MURV had separated from the dolly. This type of design would increase the weight of the dolly and add to the complexity of the main wheel assembly. The second potential solution would be to incorporate a set of rear skids that would deploy once the MURV separates from the dolly. These skids would lift the aft end of the dolly high enough that the rear wheels were no longer in contact with the ground, thereby increasing the friction force acting the skids. Revisiting the landing distance calculations done in Appendix P, a preliminary evaluation using this skid design was conducted. From this evaluation a stopping distance of approximately 200 feet or greater was reached. Therefore the use of skids alone to brake the dolly would not suffice. The final alternative considered was the use of a drogue chute located at the rear of the dolly. Using the same analysis used in Appendix P, it was calculated that a drogue chute, four feet in diameter could stop the dolly in approximately 115 feet based on the following assumptions:

1. The overall dolly weight equals 30 pounds;
2. The parachute has a drag coefficient (C_D) equal to 1.4;
3. The coefficient of friction (μ) between the wheels of the MURV and the ground equals 0.05;
4. The velocity of the dolly at the point where the drogue chute is deployed equals 109 $\frac{ft}{s}$;
5. Sea level conditions.

A drogue chute of this size can easily be mounted within a canister with a spring loaded plunger to force the chute into the airstream where it can rapidly inflate. Additionally, a drogue chute of this size should not pose any difficulty in repacking for quick reuse on another flight.

Some type of shock cord should be used in conjunction with either a KEVLAR or nylon bridle to attach the drogue line apex to an attachment point on the dolly. The purpose of this shock cord would be to help transition the load experienced by the dolly when the drogue chute became fully inflated. Additionally, the attachment point or points on the dolly may need to be reinforced. Also the opposite direction of the the vector for the drogue chute load from the attachment point(s) should intersect at or slightly below the center of gravity of the dolly. This will ensure the stability of the dolly during the deceleration after drogue chute deployment.

5.5.2.5 Acceleration Device (if required) In the event that the combined weight of the MURV and its dolly are so great that the thrust provided by the MURV's internal engine can not accelerate the MURV to takeoff velocity within the constraint of the runway length at the Jefferson Proving Ground, an additional propulsive force could be added to the dolly. This additional force could be supplied by many means such as; a small electric motor, a small gasoline engine, or a flywheel. For any one of these, a torque would be applied to the rear axle. This rear axle would be considerably more complex than the one currently envisioned. This information was provided in the highly unlikely probability that the weights for the MURV and dolly do become to great. We do not propose that the use of an additional propulsive force be considered unless the need does arise.

5.5.3 Recovery Equipment As discussed in the introduction to this section, the recovery system is made up of a rear set of skids located behind the inlet and a front mounted wheel located just aft of the nose cone bulkhead. During the launch of the MURV from its dolly, all of the above are pretracted into the fuselage so as to provide the clean aerodynamic shape required for the supermanueverability testing identified in Section 3.1. When they are deployed they form a conventional tricycle undercarriage. The following sections describe in detail the major components of the recovery system. The preliminary design work that applies to these major components was done based on the dimensions and mass properties for an early version of the selected configuration. Even though much of the analysis is preliminary many aspects of the analysis will help shed light on the design process and emphasize the features of the onboard recovery equipment.

5.5.3.1 Rear Main Skids The location of the rear skids can be calculated using knowledge of the most rearward c.g. position. "The main wheels should be located aft of the most rearward c.g. to insure that the aircraft will not tip on its tail under normal circumstances, but this point should not be too far aft; otherwise, the load on the nose wheel will be too high . . . The normal nose wheel load is in the order of 10 to 20% of the aircraft weight. A too light nose wheel load will make steering inadequate. The best compromise is to have approximately 15% of the weight of the airplane on the nose wheel at the static level attitude . . ."[76:10]. Because of the availability of a mounting location for the nose gear right behind the nose cone bulkhead, this postion was held fixed for this calculation. The static loads can be easily calculated given that the nose wheel should carry between 10 and 20% of the weight of the MURV. Once the static loads are known, the position where these loads are applied can be found by summing the moments from the loads about the center of gravity of the MURV and setting this sum equal to zero. The moment arm from the nose gear to the c.g. is known, so the only unknown is the distance from the rear skid strut to the c.g. Table reftable:loads summarizes the loads and positions for three settings of the % of takeoff gross weight (TOGW) carried by the nose gear.

Table 5.7. Landing Gear Attachment Locations

% TOGW on	Loads (lbs)		Station Location (ft)	
Nose Gear	Nose Gear	Main Gear	Nose Gear	Main Gear
10	21	189	1.88	5.49
15	31.5	178.5	1.88	5.70
20	42	168	1.88	5.94

To enable a better understanding of the potential interference that the location of the rear skids can cause, Figure 5.17 is presented to graphically depict the loads from Table 5.7. In this figure the loads are shown in respect to the most aft c.g. location. Additionallly, a preliminary inlet design is shown and the structural rails are included to show the available volume for the rear skid strut design. From this figure it can easily be seen that the location of the rear skid strut mounting locations and the point where the inlet enters the bottom of the fuselage are very close to each other. A considerable amount of analysis and design will be required to ensure that the mounting bracketry and deployment equipment for the rear struts does not interfere with the inlet. Moving the mounting location nearer to the 20% TOGW point does increase the volume available as seen in the top view of Figure 5.17. Many tradeoffs are applicable for this location. For example, the location of the front strut could be moved farther forward thus allowing the rear struts to be moved farther aft. Additionally, allowing the percent of TOGW load on the nose wheeel to increase will accomplish the same effect. Both of these have implications to the design of the front nose wheel,

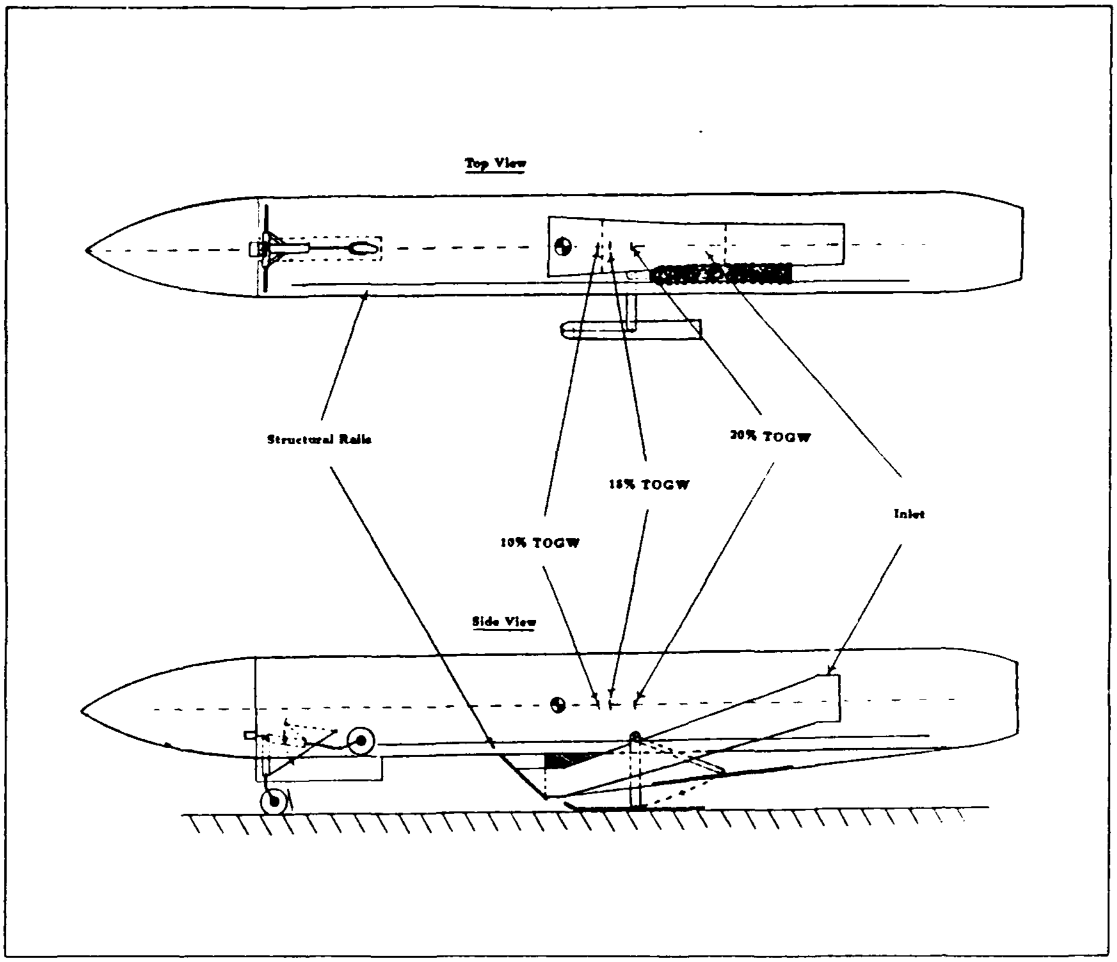
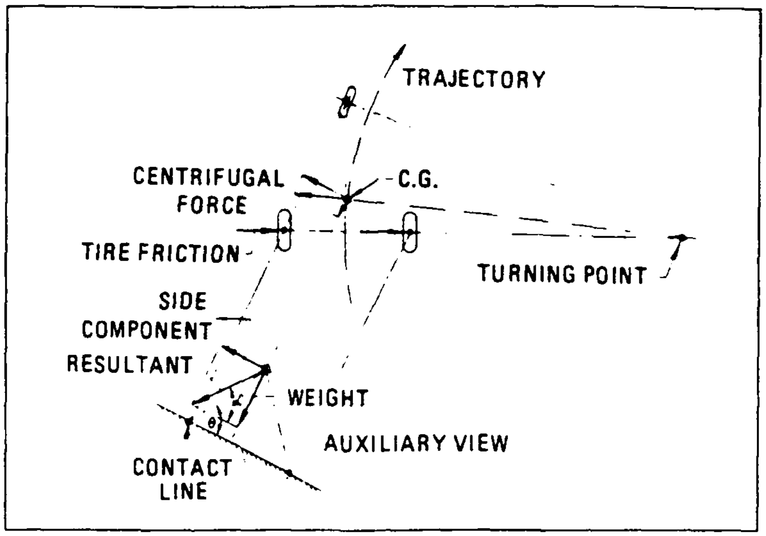


Figure 5.17. Top and Side View of the Landing Gear Attachment Locations

therefore care must be used in choosing the final rear strut mounting location. For this iteration we kept the nose wheel right behind the nose cone bulkhead and positioned the rear skid strut mount at the 20% TOGW nose load location.

Now that the longitudinal locations for the front and rear landing gears have been established the next step in the design process involves establishing the lateral positioning of the rear skids. This is required to ensure the lateral stability of the MURV. "The lateral stability is defined as the side force that would act to overturn the aircraft while in a turning trajectory as shown in Figure 5.18. The force resisting overturning is the aircraft weight at the c.g. In Figure 5.18, if the resultant force falls outside of the contact line as shown, with the strut and skids deflected proportionately, the aircraft will overturn (angle α larger than the turn-over-angle θ)"[76:13]. By using the technique



[76:13]

Figure 5.18. Forces in a Turn

described by Ladislao Pazmany, [76] the tread was calculated to be approximately 19 inches for a a marginal θ of 63°. Reducing θ to approximately 57° increases the tread to almost 22 inches. The problem with a vehicle such as the MURV is its long wheel base. Because the wheel base can not be reduced appreciably the only other parameters that can be varied are the height of the rear struts and the tread. Reducing the height will reduce the clearance for the inlet which can not be tolerated, leaving only the tread as a variable. Because of this, part of the design for the rear skids will have to include a mechanism that not only deploys the skids downward but outward as well. For a tread of 22 inches, this mechanism would have to pivot the skids out approximately 5.5 inches outward from their stored position.

Now that the position of the attachment point and the location of the skids after deployment have been identified, the next area of focus is on a method to absorb the shock induced by the landing. Unlike the use of wheels, the skids themselves will not contribute to the absorption of this shock. Many different types of shock absorption methods could be used by the MURV. However, because the weight of the landing gear is so critical this number was greatly reduced. The simplest method (also the lightest) is based on the properties of a cantilever spring. The main features of this type of landing gear are best expressed by the inventor himself, Mr. S.J. Wittman in his U.S. Patent 2,163,653 of June 27, 1939:

> One of the primary objects of my invention is to provide an airplane landing gear of an expecially simple, strong, and durable construction, and one in which the frontal area of head resistance is reduced to a minimum.
>
> Another salient object of my invention is to provide a fool-proof landing gear, which, after being assembled on the plane, requires no further attention from attendents and the like, the landing gear entirely eliminating the use of oil and spring shock struts, shock cords and the like, as now commonly employed on airplanes.
>
> A further important object of my invention is to provide a landing gear having the wheel struts thereof formed directly from resilient material and extending downward from the fuselage at an angle, and directly supporting the landing . . .

One of the biggest advances since the time when Mr. Wittman patented this type of landing gear is that of composite materials. These materials can have elasitic properties greater than steel with a weight reduction of approximately 60%. Additionally, composites have better fatigue resistance than that of high strength metals.

Because the main landing gear of the MURV is pretracted before landing, additional considerations have to be addressed. While using a composite cantilever spring provides a strong light weight strut, a method to rotate the strut downward needs to be included. Additionally,some form of shock isolation joint should be included at the attachment point on the fuselage to dampen the shock of landing and the vertical and horizontal loads during the deceleration. Both of these could be incorporated into a single unit that would rotate the strut into the deployed position and transfer the loads.

Other minor components of the rear landing gear design include:

1. Pivots on the skids that will allow the skid to remain oriented parallel to the centerline of the fuselage.

2. A method to lift the front lip of the skid so that it does not catch on any imperfections in the runway surface.

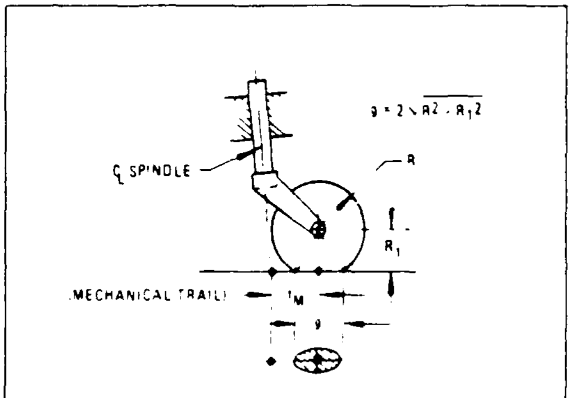
3. A device that locks the struts in the down position.

4. A sensor to verify the landing gear is in the pretracted position.

5. A sensor to verify the landing gear is in the down and locked position.

5.5.3.2 The Front Nose Wheel As mentioned in the previous section the location of the front landing gear attachment to the fuselage is approximately 19 inches back from the nose of the MURV or just aft of the nose bulkhead. The biggest advantage to this location is that additional structure will have already been added to the fuselage design so that only minor additions will be needed to beef up the structure so that landing loads can be handled. Additionally, from the previous section a load factor of 20% of the TOGW was assigned to the front landing strut during level load conditions for this iteration.

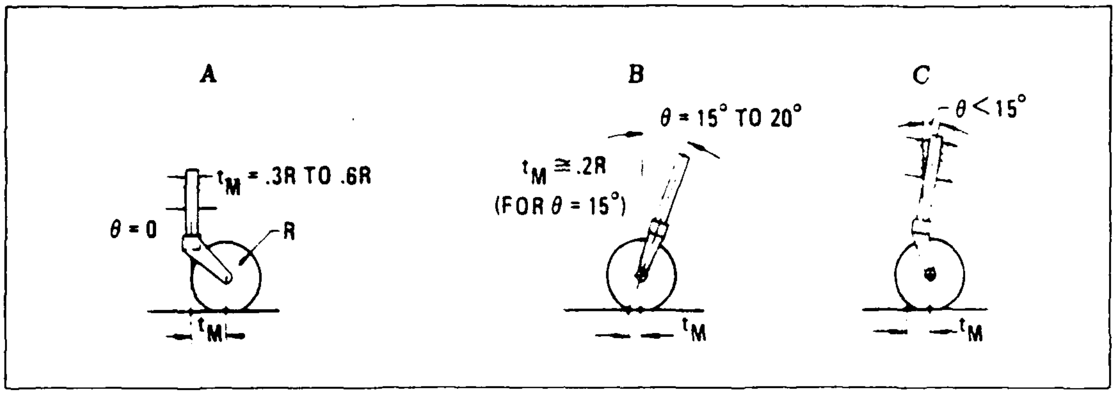
To allow the nose wheel to caster properly, the center of contact of the tire with the ground must be behind the spindle axis intersection with the ground line. This is defined as "positive trail", and is inherrently stable [76:14] (see Figure 5.19).



[76:14]

Figure 5.19. Static Trail

Three common nose gear configurations are shown in Figure 5.20. Configurations B and C would cause the nose of the aircraft to be lowered when turning and therefore are statically unstable and would require additional steering force. Configuration A is both statically and dynamically stable. It incorporates a positive trail and as such was our preferred choice for the front nose gear general configuration.



[76:14]

Figure 5.20. Nose Gear Configurations

The front landing gear can be broken into three major components; the landing strut, the wheel assembly, and the combination retraction/steering mechanism. Similar to the main struts, the strut of the nose gear must incorporate a means of absorbing the loads from the landing and during deceleration. Once the skids touchdown, a moment will be created that will increase the load on the nose strut greater than a similar aircraft with wheels on the rear struts. This is caused by the high coefficient of friction for the skids. Unlike the main struts the nose strut can not use a design such as a cantilevered spring. Instead some type of shock absorber must be used. There are quite a few different types of shock absorber designs, including internal and external helical spring, oleo-pneumatic, articulated, and many variations for each of these, available for the nose strut. For this iteration any decision on a shock absorber type was deemed unnecessary.

The design of the nose wheel assembly was focused around the requirement for steering during the deceleration. To achieve the best steering authority the nose wheel requires good ground surface contact. Typical inner tube tires provide more than adequate ground contact for the MURVs turning requirements. As with the selection of the shock absorber previously, a particular wheel design decision can be delayed.

The final component of the front landing gear encompasses the retraction and steering features of this gear. Preliminary analysis of available volume in the MURV, just aft of the nose cone bulkhead showed that the volume for the retraction of the front nose gear was minimially restricted. The simplest retraction process would have the nose gear deploy as depicted in Figure 5.21. With this process, points A and E are fixed to the fuselage and the wheel deploys down and forward. The points B' and D' represent the locations of the retraction strut in the stored position and the points B and D depict the strut after deployment. The front nose gear of the Sequoia F8L "Falco"

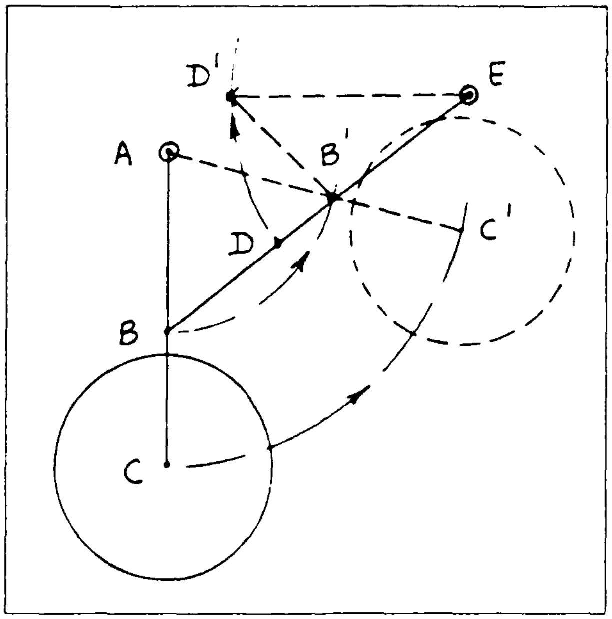


Figure 5.21. Front Nose Gear Deployment

shown in Figure 5.22 has many of the basic nose wheel requirements built into a compact package. The retraction fulcrum axis would provide the simple attachment to the fuselage and the steering arm could be easily integrated into a servo controller to provide the steering capability. We decided that this type of design was an excellent intitial design for the final nose gear design to be further refined and optimized around. Two additional components of the nose gear design are a FOD

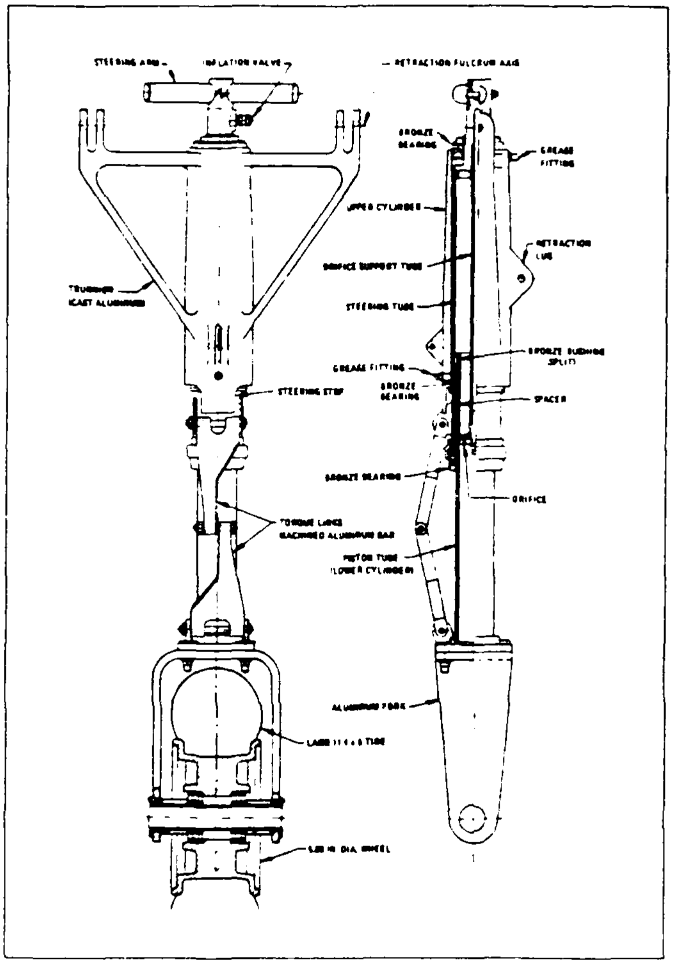


Figure 5.22. Sequoia F8L "Falco" Nose Gear

protection system and the nose gear door. The same considerations for FOD of the launch system apply to the recovery system. Because the rear main gear is located behind the inlet it does not have any FOD effect. However, the front nose gear does present a FOD potential. Two areas of the design could alleviate this problem. The first applies to the nose gear and the second applies to the engine inlet. The wheel of the nose gear causes the FOD by kicking up debris from the runway as it rotates. A solution would be to include a shield (similar to an auto mud guard) behind the wheel. This would insignificantly change the nose gear design while providing a large reduction in the probability of FOD. An alternative method to protect the inlet is to provide a screen for the inlet. This screen would be composed of the same material that was discussed for the cradle design in Section 5.5.2.3. Prior to landing, this screen would be stowed along the bottom of the fuselage. Just prior to landing it would deploy and cover the face of the inlet, thereby, providing positive control over the size of particle that could be ingested. This method provides a more positive control for FOD protection, but it is far more complicated then the first alternative.

The nose gear door can be designed in two different manners. One way would be to physically attach a section of the nose gear door to the strut and have a smaller section of the door be pivoted at the back of the nose gear wheel well. This small section has to be included because of the deflection of the nose strut during landing. The other way would be to hinge the door longitudinally along the side of the wheel well so that it remains attached to the fuselge. The hinge could be spring loaded to ensure that it is restricted from opening fully and from fluttering. Because of the simplicity of the second method and the increase in force required to open the first (due to the increase in the drag force on the section of door attached to the strut) this method was chosen to be carried into the detailed design phase.

Finally, as with the main gear design, sensors are required to verify that the nose gear is in the stowed position and that the nose gear is in the down and locked position after deployment.

5.6 Summary

The preliminary design of the launch/recovery system of the MURV has identified the best launch and recovery methods for the MURV. Based on the overall system objectives for the MURV, separate launch and recovery systems resulted. The launch system is a three wheeled dolly that supports the MURV on the ground and during the ground roll acceleration. Upon reaching liftoff velocity the MURV separates from the dolly leaving the dolly on the ground. The recovery system consists of a set of rear mounted skids with a front nose wheel. Both the skids and the nose wheel are pretracted prior to landing and therefore only require simple downward acting deployment devices. The advantages of these designs are the low weight penalty for the flight vehicle and the overall simplicity of the designs. Both of these systems are good baselines for which detailed design work can begin.

VI. Data Acquistion

The goal of this section is to define the form and functional requirements for the MURV data acquisition system, to include instrumentation, telemetry, airborne signal conditioning and interface requirements with related systems. The topics of data recording and data processing done on the ground are covered in Chapter IX of this volume. The unique requirements for MURV subsystem performance, growth capability, and modularity are emphasized. Throughout this section, theoretical information pertaining to particular subjects is given in order to facilitate better understanding of the material presented.

6.1 Data Acquisition System Objectives

The system objectives, as discussed in Volume One, Chapter II, were the basis for defining subsystem constraints and objectives. Therefore all of the general requirements for the data acquisition system were derived directly from one or more of the system requirements.

The need for an instrumentation system resulted from the experimental nature of the MURV. In order to be used for flight test, the MURV must reach and maintain desired flight conditions and be controllable within certain limits. In addition, it is necessary to gather particular experimental data for analysis. Thus, the most basic requirement of the data acquisition system is to provide appropriate information to the pilot, the flight control system, and the experimenter. This data must be provided in a usable form, with sufficient accuracy. To increase the MURV's experimental flexibility, the goal of maximizing data accuracy was established.

The system objective of flexibility, or the ability to perform a variety of experimental flight tests and achieve various aircraft configurations, required flexibility in the data acquisition system. This flexibility was best achieved by providing expansion capability and designing for subsystem modularity and the enhancement of system modularity. An example of subsystem modularity is the ability to change one component or function of the system, without replacing related hardware components, and making the change with a minimum of interface problems. Enhancing, or not restricting, system modularity refers to the ability of the data system to conform to changes in the aircraft configuration, in areas such as sensor location, equipment mounting, and cable routing.

An important aspect of sub-scale aircraft design is the limited onboard volume. Additionally, to achieve the desired aircraft performance, it was necessary to minimize the weight of onboard components. For these reasons, a design objective was established to minimize the onboard data system equipment size and weight, without reducing system performance, such as data accuracy.

The reliability of each subsystem directly affected the system reliability, therefore maximizing reliability was an important design objective for the data system. In addition, the reliability of the data/command telemetry links contributed to the reduction of risk associated with flying the MURV. The benefits and costs of component redundancy was also investigated.

The objective of simplicity in design and operation was used in specifying requirements for *both onboard and ground-based components. Since in most cases, increased complexity translated* into increased cost, the minimization of cost was also achieved.

Market availability became an increasingly important aspect of the design when decisions affecting hardware design were made. The goal was to define a design which would allow flexibility in hardware implementation. For example, if flight test aircraft commonly used a particular type of telemetry system, it was assumed that these systems and interfacing hardware would be widely available and designed specifically for the flight test mission. In such a case, that technology was closely examined for its applicability to the MURV.

The initial step in specifying requirements for the data acquisition system was to analyze and further define the data requirements generated by the flight control system, the pilot station, and the experiments.

6.2 Onboard Data Collection

6.2.1 Data Requirements The design of the instrumentation system began with the specification of certain data requirements. These requirements were generated from analyses done for aircraft performance determination, experimental needs, flight control system requirements, and remote cockpit requirements. Discussions of the first three analyses were given in previous sections, and the remote cockpit, or pilot station needs are defined in Chapter VIII. It is likely that additional test-specific data will be required at some future date. This was accounted for in the built-in expansion capability of the data acquisition system. When a more significant impact on the data system was foreseen, the effect of those additional data requirements was estimated.

Table 6.1 lists the set of baseline parameters to be measured onboard, the range of values expected, the intended uses of each parameter and the sensor location. All of the parameters listed will be recorded and possibly used for experimental analysis, in addition to the uses listed.

The minimum and maximum values listed here are preliminary, and were derived from investigation of similar aircraft systems and initial performance calculations. Although the engine data were based on a single engine design using the Teledyne engine, Model 312, these approximate values would also apply to Teledyne Model 320 (Chapter III). All parameter ranges must be further refined as the MURV design progresses.

The particular use of each parameter was needed to specify the required data flow, which would affect telemetry capacity, and recording and processing capability. In addition, the data accuracy requirements will be directly derived from the intended use of the data.

Sensor location is important in determining physical interface requirements onboard the MURV and the environmental conditions the transducers will see. Accelerometers and position and rate gyros should be located as close to the center of gravity of the aircraft as possible, to reduce the amount of correction necessary. The air data (pressure, temperature, angle of attack, angle of sideslip) probe will be located on a fuselage nose boom, as specified in Section 6.2.3. Transducers used to convert the air pressure data to an analog (or digital) signal must be located close

Table 6.1. Parameters Identified for In-Flight Measurement

Parameter	Units	Range Min	Range Max	Intended Use	Sensor Location
Normal Acceleration	G	-3	9	AP†, FCS‡	A/C c.g.
Longitudinal Acceleration	G	-3	3	AP	A/C c.g.
Lateral Acceleration	G	-3	3	AP, FCS	A/C c.g.
Angle of Attack	degrees	-30	90	AP, FCS, RC⋆	Nose Boom
Angle of Sideslip	degrees	-45	45	AP, FCS, RC	Nose Boom
Pitch Angle	degrees	-90	90	FCS, RC	A/C c.g.
Roll Angle	degrees	-180	180	FCS, RC	A/C c.g
Pitch Rate	deg/sec	-100	100	FCS	A/C c.g.
Roll Rate	deg/sec	-720	720	FCS	A/C c.g.
Yaw Rate	deg/sec	-30	30	FCS	A/C c.g.
Static Pressure	psia	0	15	FCS, RC	Nose Boom
Total Pressure	psia	0	20	FCS, RC	Nose Boom
Air Temperature	Deg C	-55	85	FCS, RC	Nose Boom
Magnetic Heading	degrees	0	360	PS	Equip. Bay
Control Surface Position	degrees	*	*	FCS	C.S.
C.S. Hinge Moments	in-lbs	*	*	RC	C.S.
Engine RPM	RPM	20,000	72,000	FCS, RC	Engine
Engine Fuel Flow Rate	lbs/hr	0	150	FCS, RC	Fuel Sys.
Engine Inlet Temperature	Deg C	-55	85	FCS	Engine
Engine Exhaust Temperature	Deg R	510	2100	FCS	Engine
Engine Inlet Pressure	psia	*	*	FCS	Engine
Engine Exhaust Pressure	psia	*	*	FCS	Engine
Video Camera Picture	N/A	N/A	N/A	RC	A/C Nose

†Aircraft Performance Calculations
‡Flight Control System
⋆Remote Cockpit
*To be determined in a follow-on study

to the pressure source to reduce the effects of pressure lag due to pneumatic hose length. Engine RPM measurement is provided for on all of the engines that were considered. Fuel flow rate will be obtained from a flow meter in one of the fuel lines.

Engine temperatures and pressures will be gathered from sensors which are integrated into the engine. The engine data included in Table 6.1 are the baseline requirements and will be used to establish data flow, telemetry, and signal processing and recording requirements. Expansion capability will be provided in all of these areas to account for unforeseen data requirements.

Further characterization of each parameter is needed before an instrumentation system can be designed. This additional information includes:

accuracy Data accuracy requirements drive all of the components of the data acquisition system and are dependent upon the desired use of the measurement, e.g., flight control system and performance calculations. Usually given as a percentage of the full scale reading, or in number of units, when the full scale reading is known. Typically, 1–5 percent accuracy is achievable.

resolution Resolution is defined as the smallest measurable change in the parameter and is driven, in part, by the accuracy requirement. Usually given as a percentage of the full scale reading. Becomes important when continuous data is sampled.

frequency response The frequency content of a data signal must be estimated in order to select the appropriate transducer and establish signal conditioning requirements, such as sampling rate.

The parameter information listed above drives the complexity and cost of several components of the system, including transducers, signal conditioners, analog-to-digital converters, and the telemetry system. In this analysis, only the functional requirements for the data acquisition system were defined. Actual component selection was not done. In cases where specification of the above quantities was needed to make order-of-magnitude estimates, available data from other test programs was substituted.

A number of discrete (2-position) parameters were also required by both the flight control system and the remote cockpit. These include:

- Telemetry uplink/downlink status
- Critical sensor status
- Landing gear position
- Landing gear locked/unlocked
- Video camera status

- Backup flight control system engaged indication
- Power supply status

The discrete parameters are required primarily for reliability and safety considerations. These will be discussed in more detail in Section 6.6.

6.2.2 Transducers A transducer is a device which converts a non-electric signal into an electric current or voltage that reproduces the information in the signal. Improper selection of transducers can cause degradation in the data of interest, to the point that it is rendered useless to the experimenter. Selection of a transducer is primarily affected by the properties of the parameter being measured, the desired accuracy of data, environmental conditions and integration with the telemetry system.

Actual transducer selection was not made in this thesis effort. Transducer selection is based on a more detailed characterization of the parameters than is available at this time, therefore is left for a follow-on effort. Preliminary guidelines for transducer selection are provided in Appendix O.

The measurement of air data, such as air pressures, temperature, angle of attack, and angle of sideslip, involves much more than selection of transducers with necessary properties. Their measurement requires an interface with the exterior structure of the aircraft and is affected by Mach number, angle of attack, yaw angle, and control surface positions. The supermaneuverability role and relaxed stability requirement of the MURV demand high data accuracy for these parameters, and at the same time, make their accurate measurement especially difficult. For these reasons, a preliminary investigation into methods for measuring airspeed, altitude, angle of attack and angle of sideslip was done.

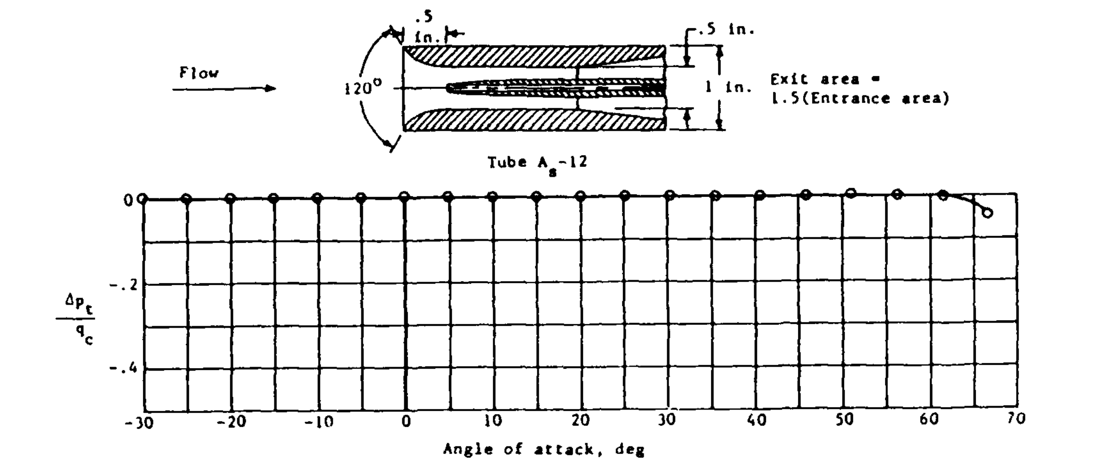
6.2.3 Air Data Measurement The supermaneuverability role of the MURV places demanding requirements on the air data system. Errors in pitot and static pressure and flow direction measurements (angle of attack and angle of sideslip) increase with angle of attack. In addition, the severe maneuvering required to obtain those high angles of attack significantly increases the effects of pressure transmission lag.

Accurate air data measurement requires calibration for pitot pressure, static pressure, air temperature, angle of attack and angle of sideslip [62]. The calibration is typically valid only for a specific sensor design and installation. Because pressure and flow measurement is so dependent upon the flow field around the aircraft, any significant MURV configuration changes would affect the calibration of the air data sensors. For this reason, an effort was made to select an arrangement that would be least sensitive to aircraft configuration changes. An additional concern would be in reducing the requirement for sensor relocation due to major aircraft changes, such as a tail or wing replacement.

The position for air data sensors that would be least affected by aircraft changes is a nose-mounted boom. Although desirable for modularity concerns, further investigation was required to verify that accurate data could be obtained with an aircraft nose-mounted air data system. A preliminary investigation of sensors and sensor locations suitable for high angle of attack flight at subsonic speeds was done to determine the impact on the flexibility of the MURV data acquisition system. The research focused on the measurement of total pressure, static pressure, angle of attack and angle of sideslip.

6.2.3.1 Total Pressure Measurement A pitot tube aligned with the air flow and operating at subsonic speeds will register total pressure correctly if the tube is located away from any boundary layer, wake, or engine effects [32:6]. The major concern of total pressure measurement is the change in sensed pressure when the pitot tube is not aligned with the air flow. This sensitivity to flow alignment can be controlled by pitot tube design. Standard fixed pitot tubes, such as the one shown in Figure 4.1, can accurately measure pressures over only a small range of angles of attack (within $\pm 15°$).

Accurate total pressure measurement can be obtained for large inclination to air flow with pitot tubes equipped with a pivot and vanes to align the tube with the airstream, or by a shielded fixed pitot tube (Kiel-type). The total pressure error as a fraction of impact pressure for a shielded pitot tube can be seen plotted versus angle of attack in Figure 6.1. Identical performance should be seen for sidelslip angle when the pitot tube design is symmetric.

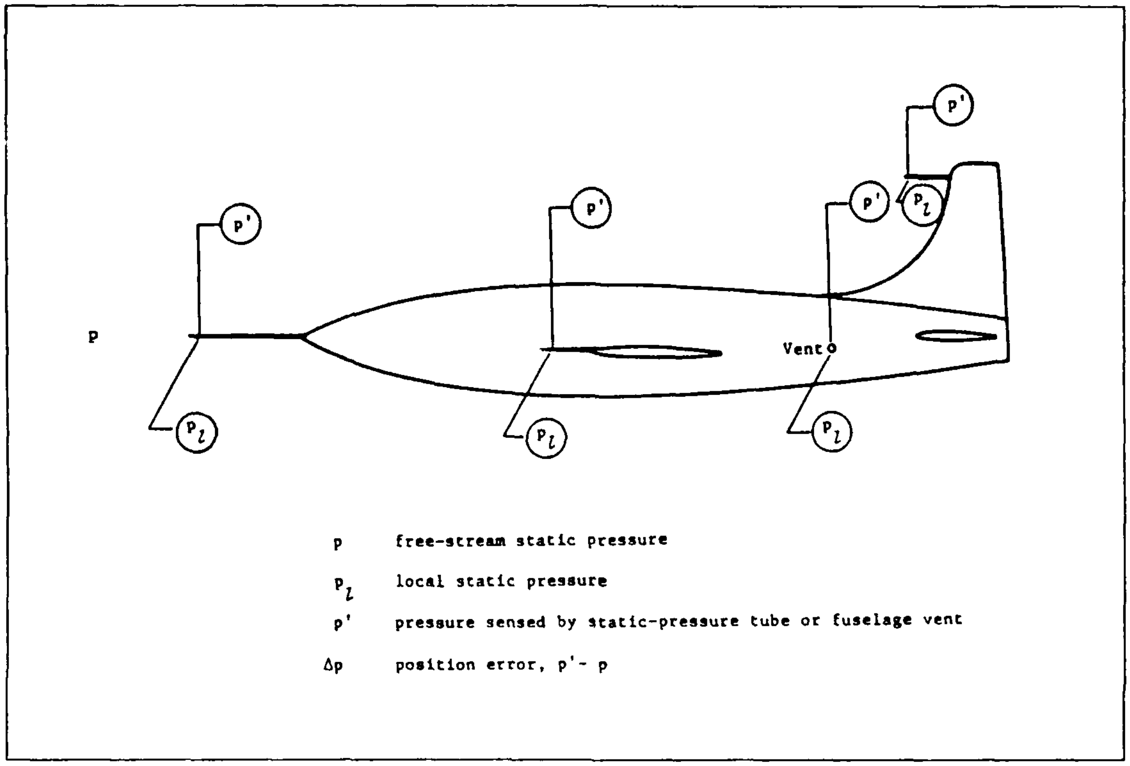


[32:39]

Figure 6.1. Variation of Total Pressure Error with Angle of Attack for a Shielded Tube

Shielded and pivoted tubes are not commonly used in flight test aircraft however, because due to the relatively large flow field around the tube, they cannot be used in a pitot-static tube arrangement. Static pressure measurement must therefore be made at another position on the aircraft. It is not uncommon however, to use these high-accuracy pitot tubes to calibrate the standard aircraft pitot system [62:21].

6.2.3.2 Static Pressure Measurement Static pressure measurement error is largely due to the location of the static port on the the aircraft. This is commonly referred to as position error. Reference [32] presents an analysis of position error based on sensor location on an aircraft. The orifice positions investigated include those mounted in tubes positioned ahead of the fuselage nose, vertical tail, or the wing leading edge, near the wing tip, strut-mounted horizontal tubes attached to the fuselage or wing, and fuselage vents, as depicted in Figure 6.2.



[32:46]

Figure 6.2. Diagram of Types of Installations for Static Pressure Measurement

The conclusion from two sources is that a fuselage nose installation is the best choice for static pressure accuracy at supersonic speeds. However, for the subsonic range, which is our area of interest, the wing tip or fuselage vent installations typically provide better static pressure measurement. A nose-mounted boom should extend a minimum of 1.5 fuselage diameters ahead of the nose [32:95]; [28:839]. Quantitization of the position error through flight calibration procedures can reduce the adverse effects due to sensor location. Additionally, a water tunnel test using a scale model of the MURV should be done to determine flow patterns before final positioning of any air data sensors.

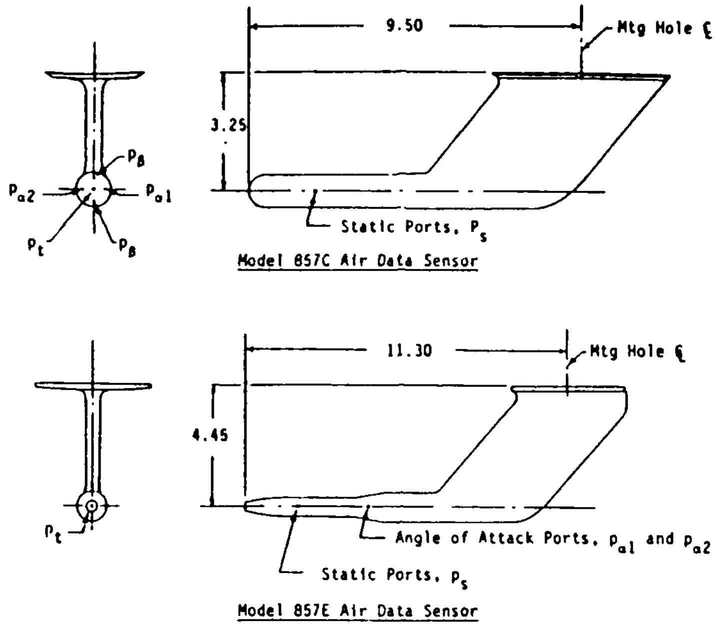
6.2.3.3 Flow Direction Measurement The error in angle of attack angle of sideslip measurement is affected by sensor design, sensor location, and for accelerated maneuvers, the by aircraft pitch and yaw rate, the deformation under load of the aircraft and nose boom, and pressure transmission lag [62:44]. Two basic types of flow direction sensors available are:

1. Free-floating vanes
2. Differential pressure probes

The free-floating vane can be subject to flutter and unsteady readings at high angles of attack. However, the damping introduced in the pivot to prevent this oscillation also inhibits accurate angle of attack measurement for high pitch rates. The differential pressure probe requires an arrangement of pressure orifices at the end of a tube. The difference in pressures at these orifices indicate the flow direction, and therefore the angle of attack or sideslip. High pitch rates increase the measurement error for these devices also, due to pneumatic lag. This effect can be lessened by positioning the pressure transducer close to the sensor location.

6.2.3.4 Integrated Sensors Integrated sensors, or multiple output air data sensors, provide pitot pressure, static pressure, angle of attack and angle of sideslip with a single probe. They work on the concept of differential pressure at various pressure ports on the tube. Reference [35] describes two multiple output sensors that are used in concert with a digital air data transducer. These sensors can be seen in Figure 6.3.

This particular configuration allows direct connection of the transducer to the base of the air data sensor to reduce pressure lag effects on static pressure and flow direction measurement, and can provide accurate pitot pressure and angle of attack data for high angles of attack. Static pressure position error can be compensated for, based on the results of flight calibration. With the use of a digital transducer, the integrated air data sensor can correct for its own errors as a function of Mach number and angle of attack [35].



[35:383]

Figure 6.3. Two Typical Multiple Output Air Data Sensors

6.2.3.5 Recommended Air Data System An integrated air data probe, placed at least 1.5 fuselage diameters ahead of the fuselage, with a digital transducer is recommended for use on the MURV based on the following considerations:

- Pitot pressure, static pressure and angles of attack and sideslip can be obtained using a single probe.
- Use of a fuselage nose mounted boom reduces the effects of aircraft configuration changes.

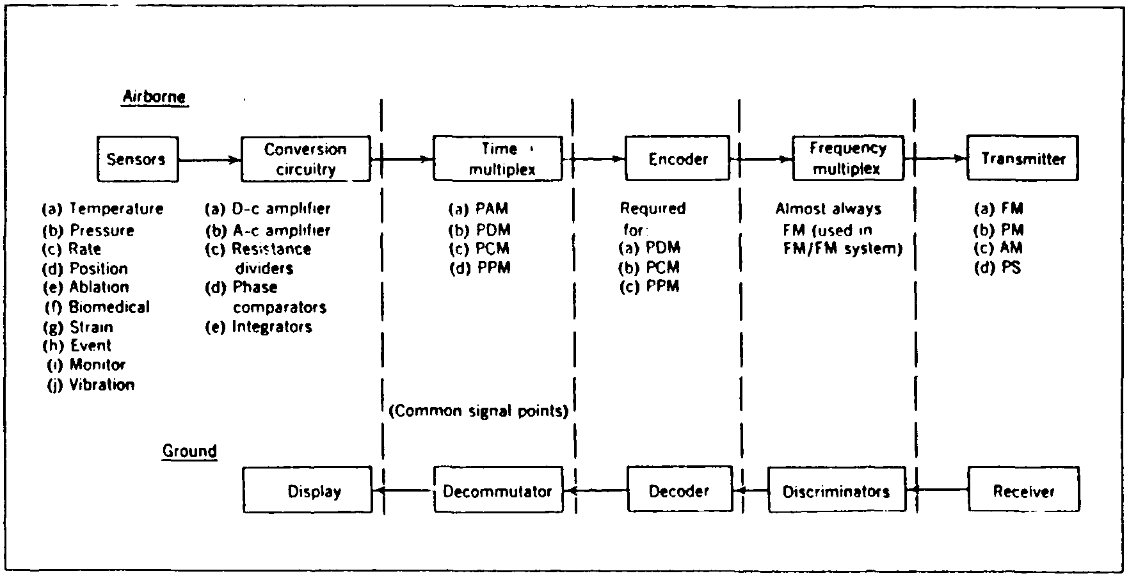
- High accuracy results can be obtained for high angles of attack and various Mach numbers by self-contained error compensation.
- Location of the transducer near or directly connected to the sensor reduces the effects of pneumatic lag.

6.3 Telemetry System

A major function of the data acquisition system was to transmit to the ground station the video camera image, all of the data required by the pilot, the data inputs to the ground-based flight control system, and the data that was to be recorded on the ground. (Ground recording was chosen over onboard recording for a number of reasons—see Chapter IX.) Included as a part of the data acquisition system was the telemetry uplink for command and control of the MURV.

Telemetry, as discussed in this paper, is the transfer of information via a radiowave carrier to a remote location for recording, display or actuation of a process [31:1]. A schematic of a typical telemetry system that provides a downlink to the ground is shown in Figure 6.4. The major components of the onboard telemetry system are signal conditioners, encoding modulators, multiplexer, rf (radio frequency) modulator and transmitter. The ground-based components consist of a receiver, demodulator, demultiplexer, and decoder. Encoders and decoders are required only for digital transmission systems. For the uplink to the aircraft, the transmitter, modulators and multiplexer are ground-based and the receiver system is onboard the aircraft. After the modulated, multiplexed radiowave is transmitted, it is received, demodulated, demultiplexed, and decoded to retreive the original information from the transducers.

Design or selection of a telemetry system can be accomplished based on the characteristics of the components shown in Figure 6.4. In this section, common telemetry design considerations are discussed in order to provide a background for the specification of a telemetry system for the MURV. The selection of a telemetry system type was based on the subsystem objectives defined in



[31:6]

Figure 6.4. Typical Telemetry System Components

Section 6.1.

6.3.1 Modulation Transducer signals usually cannot be transmitted directly as radiowaves. Therefore a carrier wave, with appropriate frequency, is modified to represent the information in the signal. This process is referred to as modulation, or the systematic alteration of a carrier wave in accordance with a message. In addition to matching a signal to a transmission medium, modulation can be used to reduce noise and interference and to prepare signals for multiplexing [18:6].

The two basic types of modulation are continuous-wave modulation and pulse modulation. In continuous modulation, the parameters of a sinusoidal carrier (amplitude, phase, frequency) are controlled by the signal voltage. In pulse modulation, the parameters of a pulse carrier (amplitude, duration) are controlled by the signal voltage [98:1-92]. Pulse modulation is well suited for a discrete signal, however, it can be used with a continuous signal with the use of a sampling process. Some general modulation methods that are most common to flight testing are discussed below.

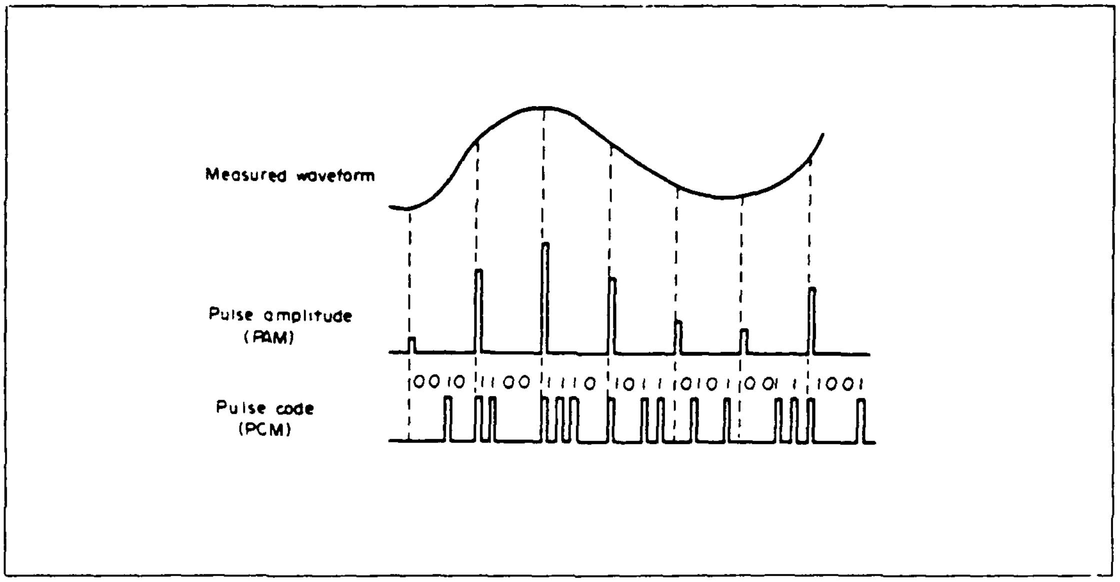
The most common continuous-wave modulation methods are amplitude modulation (AM), frequency modulation (FM), and phase modulation (PM). Frequency and phase modulation are the only continuous-wave methods commonly used in aircraft telemetry systems. A brief description is provided below.

Frequency Modulation A technique in which the instantaneous frequency of the carrier wave is varied in accordance with the modulating signal is called frequency modulation. In frequency modulation, the amplitude of the demodulated signal is proportional to the frequency deviation, therefore one can increase output signal power by an increase in the deviation [18:222]. This is commonly referred to as wideband noise reduction, because the reduction in signal-to-noise ratio comes as a result of increased bandwidth.

Phase Modulation PM is a special case of FM, where the signal controls the carrier phase, instead of the carrier frequency. Like FM, PM is also a wideband modulation method, but has less effective noise reduction properties.

The most common types of pulse modulation, or coding modulation, used in telemetry systems are pulse amplitude modulation (PAM) and pulse code modulation (PCM). A representation of these methods in the time domain can be seen in Figure 6.5. For all pulse modulation methods, a continuous signal must be sampled, and the value of each sample is represented in various ways in the modulated signal

Pulse Amplitude Modulation PAM is the simplest pulse modulation technique and is frequently the initial step to obtaining other pulse modulations. The modulated signal is obtained by periodically sampling a continuous transducer output. This generates a train of pulses which are continuous in amplitude and discrete in time. PAM is more susceptible to noise than other pulse modulation techniques.



[73:38]

Figure 6.5. Pulse Modulations

Pulse Code Modulation PCM consists of converting continuous analog information into binary code. The process usually involves conversion to PAM; then the magnitude of each PAM pulse is encoded in a sequence of zeros and ones. These processes are referred to as sampling and quantization. The information capacity can be increased simply by increasing the number of binary digits per word. A PCM system is the most common type of telemetry system used in flight testing and will be discussed in more detail later in this section.

In telemetry systems, at least two modulation processes are used in cascade. The first, submodulation, is generally required before the signals can be multiplexed, or combined into one signal. The second modulation process is required to match the output signal of the multiplexer to the rf channel, is refered to as rf modulation [98:1-96], and is accomplished by the transmitter. Rf modulation is typically FM because it is less affected by interference than other continuous modulation methods. Submodulation can be accomplished by either continuous or pulse modulation techniques. A proper comparison of submodulation methods requires an understanding of multiplexing, therefore some theory on multiplexing is covered in the following section.

6.3.2 Multiplexing Multiplexing combines two or more signals into a composite signal that can be transmitted over a single radio channel, or recorded on a single tape track. Two different multiplexing techniques, time-division and frequency-division multiplexing, were considered for the MURV telemetry system.

Frequency-division multiplexing involves sending the signals in a parallel configuration, in which the frequency bandwidth of the transmission channel is shared by all of the multiplexed signals. Frequency-division multiplexing requires that several transducer outputs modulate individual subcarriers. The modulated signals are then summed. Based on the subcarrier frequencies chosen, the modulated signals are assigned a slot in the frequency domain. The multiplexed signal is usually then modulated on an rf carrier. After the signal has been received, the modulated signals are separated by a bank of bandpass filters in parallel. The number of data channels that it is possible to multiplex in this way is limited for a fixed transmission bandwidth. The most common frequency-division multiplexing system used in flight testing is an FM/FM system. Transducer signals are frequency modulated onto subcarrier frequencies which are then multiplexed and frequency modulated onto an rf carrier.

Time-division multiplexing, or commutation, is generally used with pulse modulation methods, and consists of dividing the time domain between signals. Commutation is accomplished by a rotary switch, or commutator, that samples the incoming signals at a fixed frequency and sequences them in a single pulse train. At the receiver end, the signals are decommutated by a synchronized switch. The sychronization of the switch is accomplished by inserting sychronization signals into the pulse train. In commutation, there is more flexibility in the number and types of channels that can be multiplexed. Techniques such as subcommutation and supercommutation further increase the flexibility of time-division multiplex systems.

Supercommutation and subcommutation are techniques for tailoring a time-division multiplex system based on sensor sampling rate requirements. Supercommutation increases the sampling frequency of a channel by sampling the data signal more than once per frame. This results in a decrease in the possible number of data channels. Subcommutation involves sampling a particular data signal less than once a frame, allowing more than one data channel with low sampling rate to share a position in a frame. This increases the number of channels that can be transmitted.

6.3.3 Comparison of Telemetry Systems It was stated previously that the most common telemetry systems used in flight testing would be closely considered. The types most commonly used are PCM/FM, PAM/FM, FM/FM, and hybrid systems combining PCM and FM/FM [98]. The benefits and disadvantages of each were examined with respect to the data subsystem design objectives. These included maximizing data accuracy, system flexibility, reliability, simplicity and market availability and minimizing onboard size, weight and system cost.

PCM has been described as the most efficient and reliable modulation method. However, the advantages of PCM are not clear-cut in every application, because other types of telemetry systems are still in use.

PAM/FM PAM/FM uses PAM for submodulation and FM for rf modulation. PAM systems are used in several Navy missile programs, where the advantages of low complexity and small size are important [91:16]. Reduced complexity, cost and size were objectives of the data acquisition system, but it was doubtful if a PAM system could produce the data accuracy required for the MURV. Because of the many sources of error, PAM systems are usually rated at between 2 and 5 percent accuracy [96:3.6-18].

FM/FM As described earlier, FM/FM systems use FM for subcarrier modulation and FM for rf modulation. Like PAM systems, the accuracy of data transmission is relatively low in comparison to PCM. In addition, a system that relies on frequency-division multiplexing is limited in the number of channels it can transmit. At the transmission frequencies usually allocated to aircraft flight test telemetry systems, 1435–1535 MHz, the average number of data channels available for one link is 21 [98:1-104]. One rf link would then be insufficient for the MURV data requirements. Advantages of FM/FM systems are that it is a well-proven technology, and has the capability to transmit wideband signals at high data rates.

PCM/FM One of the most important advantages of a PCM system is its resistance to noise and distortion. The demodulation task consists of differentiating between only two levels of the signal, making it less sensitive to errors due to noise. Error correction and detection coding further extend the error-free range. Sacrifices for the high accuracy of PCM come in the areas of cost and complexity. However, due to the ever-decreasing cost of improved computer capability, the implementation of PCM has become more affordable. In fact, due to its superior noise and distortion performance, PCM has become the norm for advanced flight test systems. The use of PCM also eliminates the need for analog-to-digital conversion at the ground station, thus simplifying integration to the digital flight control system and digital processing and recording.

Hybrid Systems There have been difficulties in the past in manufacturing commutators and decommutators with sampling rates high enough to accomodate a number of large bandwidth parameters, such as high frequency vibration [98:1-98]. Requirements for high accuracy and a large number of data channels, in addition to high data rates, may require the use of a hybrid PCM and FM/FM system. Hybrid systems involve the modulation of separate subcarrier frequencies with analog (high frequency) and PCM multiplexed signals. These subcarrier frequencies will then be used to modulate an rf carrier.

Interviews with several manufacturers of telemetry systems indicated that PCM equipment with transmission rates as high as 1×10^6 to 10×10^6 bits per second were currently in production [101]; [69]. Based on the preliminary calculations of data rate done in Section 6.3.5, a 1 MHz PCM system should be more than sufficient to accomodate the MURV data requirements, without a need for a hybrid scheme.

6.3.4 Transmitter and Receiver Requirements Transfer of the onboard data to the ground and the command/control signals to the MURV is accomplished through the ground-based and airborne transmitters and receivers. The functions of a transmitter are to generate a high frequency carrier wave, amplify the carrier wave, modulate (usually FM) the carrier with the information signal, amplify the modulated signal, and couple the signal with the antenna to be radiated into the atmosphere [13:228].

In flight test, superheterodyne receivers with plug-in modular components are most commonly used. These types of receivers are most versatile, because they can be used with different telemetry systems by choosing the proper tuner, intermediate frequency filters, and rf demodulators [98]. The tuner allows reception of only the radio frequencies of interest. The intermediate frequency (IF) filters shift the modulated signal from the transmission frequency to a lower frequency for demodulation. Demodulation, or detection, is the process of separating the information signal from the carrier wave. It may be desirable to equip the receiver with the capability for postdetection or predetection diversity combining. The advantages of this technique are discussed in Section 6.3.4.2.

Relatively high transmission frequencies are required for two reasons, to reduce antenna length and to provide sufficient data bandwidth. IRIG (Inter-Range Instrumentation Group) Standard 106-86 [97:2-1-2-5] addresses three telemetry frequency bands:

P-band (215–260 MHz) Allocated to fixed mobile services.

L-band (1435–1540 MHz) Allocated to government and nongovernment aeronautical telemetry use, "for flight testing of manned or unmanned aircraft, missiles, or their major components."

S-band (2200–2300 MHz) Allocated to government fixed, mobile and space research services.

As seen above, L-band is the appropriate frequency band for the MURV operation. Transmittal frequencies must be assigned by a local authority, along with the necessary bandwidth restrictions.

Narrowband telemetry channels provide bandwidths of 1 MHz or less. Mediumband and wideband channels provide bandwidths of 1–3 MHz and 3–10 MHz, respectively [97:A-6].

The rf bandwidth of the transmitted signal is determined by the rf carrier deviation, the baseband bandwidth (the band of frequencies occupied by the multiplexed signal before it is used to modulate the rf carrier), and the modulation technique. Nyquist showed that the required bandwidth for transmission of binary data is one half of the data transmission rate, however, in practice PCM transmission requires between one and two times the data rate [96:3.0-2,3.0-3]. A preliminary estimate of the data rate required for the PCM downlink and uplink is calculated in the following section. Based on those results, it is clear that the PCM transmissions will require no more than a narrowband channel.

Video signals require an rf bandwidth of 3–6 MHz, therefore a separate video transmission channel will probably be required. With appropriate transmitter/receiver systems, it is possible to provide two-way communication on the same telemetry channel. One narrowband telemetry channel may be sufficient in bandwidth for both the uplink and data downlink, however, to avoid potential cross-talk between channels, we recommend separate uplink and downlink transmission. Because MURV operation requires telemetry on more than one radio frequency, care must be taken to ensure proper channel spacing [97].

6.3.4.1 Antenna Requirements An antenna is a specially designed conductor that accepts energy from the transmitter and radiates it into the atmosphere. The antenna at the receiving station (airborne or ground) intercepts the transmitted energy from the radio waves and conducts it to the receiver.

The types of antennas generally used for telemetry fall into four major categories—the relatively low cost omni-directional antenna, the fixed directional antenna (corridor operation), and single axis and two axis tracking systems. Antenna selection is based on the receiving antenna gain, directivity, beam width, and price. A paper written in 1980 estimates that omni or fixed medium gain antennas can be obtained for between \$1,000 and \$3,000, the single axis tracking antennas for between \$40,000 and \$60,000, and a two axis tracking system for more than \$100,000 [93].

The single-axis tracking antenna is best for full-scale flight tests [91:50], however, MURV flight test objectives may not require the extended range normally needed in full-scale flight test. This decreased range may allow selection of a less costly omni- or fixed directional antenna.

Section 8.3.2.4 defines a requirement to supply the remote pilot with aircraft position information to allow beyond-visual-range flight. Potential solutions to this involve the use of single-axis and two-axis tracking systems [81]. This may be the most economical solution to both the signal transmission and aircraft position determination problems. However, if a tracking antenna is not required for transmission reasons alone, then a trade-study will be necessary to justify the increased expense of a tracking antenna versus alternative methods for position determination. An attempt should be made to control other design variables in order to avoid the requirement for a tracking antenna, due to the large cost differential. A decision to fly only at Jefferson Proving Ground, which is approximately 7 by 18 miles, could possibly allow use of a fixed directional, or corrider antenna.

6.3.4.2 Diversity Combining Diversity signal combining is the process of "adding" two or more independent signals to produce a stronger signal than any of the independent signals. The most common type of diversity combining is polarization diversity. This involves combining two orthogonal polarizations, which contain all of the available power. Another common type of diversity is space diversity, which requires two or more separate antennas. In this case, the signals from the individual antennas are usually selected, rather than combined [96:5.1-1,5.3-1]. Diversity combining provides improved signal-to-noise ratio and more reliable reception.

This concept was used in the HiMat and a number of other large-scale flight test programs and should be considered in the selection of a transmission system and antenna.

6.3.4.3 Communications Reliability The effect of a complete carrier signal loss for either the uplink, data downlink, or both if they use the same rf carrier, would be catastrophic for the MURV. Provisions for this must be considered in the data acquisition system, flight control system and remote cockpit designs. Identification of loss of signal could be accomplished by a carrier-operated relay in both the airborne and ground receivers. Loss of either carrier would generate an indication to the pilot and the computer(s). Automatic relay switching to backup ground equipment could be accomplished at this time, or in the case of a total system failure, transfer of control to the onboard backup flight control system. Redundant onboard telemetry equipment would be desirable. The feasibility of this solution must be further analyzed after a more complete specification of fixed equipment and payload weights.

6.3.5 Data Rate Determination An important step in preliminary sizing of a PCM telemetry system's capability, consists of determining the required data rate, or speed at which bits are transmitted. The required data rate is a product of the data sampling rates, the word length associated with each parameter, and the number of data channels required. In this section, an order-of-magnitude calculation is done, using preliminary values for sampling rates and word length, in order to obtain an estimate of the required data transmission rate.

6.3.5.1 Sampling Rate A fundamental question that needs to be answered when designing a digital data acquisition system is, "How fast do we need to sample?" Nyquist's sampling theorem states that "a signal that is ideally band-limited can be perfectly reconstructed from samples taken at a uniform rate equal to or greater than twice the highest signal frequency" [91:41]. In practice, since no signal is ideally band-limited, it is necessary to sample at rates higher than the Nyquist sample rate. Various sources recommend minimum rates anywhere from 4 to 10 times the highest signal frequency, along with pre-sampling filtering to reduce aliasing effects.

In order to establish the minimum sampling rate of each parameter, the frequency content of those parameters must be known. Determination of that information prior to flight testing is based on knowledge of the aircraft performance, such as roll rate capability, control surface actuator properties, aircraft response time and structural vibration properties, and on experience with other data gathering systems. Since the flight performance data and other detailed analyses for the MURV were incomplete, it was not possible to quantitatively establish bandwidth information for every parameter. An appropriate approach would be to evaluate data systems on other aircraft with similar flying qualities and estimate sampling rates based on them. Futher refinement could be done as the MURV design evolves. That approach was not taken in this design effort due to the requirements of the flight control system.

Another factor important to sample rate selection is the desired accuracy of the data. Since the central flight control system relies on telemetry information, its cycle times affect the required sample rates, and therefore data transmission rates. These rate requirements may be more demanding than for a typical flight test aircraft which does not rely on the telemetry link for command and control. Sampling rate as a function of flight control system requirements was discussed in Section 4.3.2. The sample rates generated by that analysis were used in the sample calculation done for data rate.

5.3.5.2 Resolution The process of digitizing continuous data involves three steps — sampling, quantizing, and coding. Sampling generates a train of pulses of continuously variable height, such as a PAM signal. In order to digitize that signal, the sample values must be rounded off to the nearest discrete level, or quantum level. Each quantum level can be represented by a digital code of n bits, where the number of quantum levels, δ_r, is defined by

$$\delta_r = 2^n$$

The number of bits in a coded word defines the resolution of the digitized signal. The resolution of a parameter was defined earlier as the smallest measurable change in a parameter, usually represented as a percentage of the full scale value. For digital code, the resolution is $1/2^n$.

There are errors associated with both the sampling and quantization processes. Quantization error is dependent on the number of bits per word, n, and the round-off or truncation technique. To achieve the desired data accuracy, it is necessary to specify word lengths for each parameter. It is common to measure some parameters more accurately than others, thereby transmitting data words of different lengths. Data words transmitted via telemetry systems typically vary from 8 to 16 bits. Again, since the data accuracy requirements have not been fully defined, one can only do a preliminary selection of word length. The resolution recommended for the for the flight control system was a minimum of 12 bits per word. This number will be used in the sample calculations for all PCM data.

6.3.5.3 Data Channels Determination of the number of data channels required begins with a list of parameters, both proportional and discrete. Proportional data require PCM words that are defined by a particular number of bits. Discrete parameters require only one bit that signifies an on/off condition, such as a switch position, or a system status light. Therefore, 12 discrete parameters can be transmitted in a single 12-bit word.

The parameters that are measured onboard and are transmitted to the ground station, with the exception of the video camera picture, are listed in Table 6.2. The sampling rates of 55 and 220 Hz were dictated by estimation of flight control system requirements. For the data not required by the flight control system, the sampling rate of 55 Hz was used. Sampling at that rate will not be required for slowly changing parameters, such as temperatures, but will suffice for this preliminary calculation. The number of control surfaces selected was 11, which was based on the MURV-1 design presented in Section 2.4.1.

There are 25 12-bit parameters that are sampled at 220 Hz, and 24 12-bit parameters that are sampled at 55 Hz. This leads to a total of:

$$\begin{aligned} \text{bit rate} &= 25 \text{ samples} \times 220 \text{ samples/sec} \times 12 \text{ bits/sample} \\ &\quad + 24 \text{ samples} \times 55 \text{ samples/sec} \times 12 \text{ bits/sample} \end{aligned}$$

Table 6.2. Data Rates Required for the Data Downlink

Parameter	Resolution (bits per word)	Sampling Rate (Hz)	Product
Normal Acceleration	12	220	2640
Longitudinal Acceleration	12	220	2640
Lateral Acceleration	12	220	2640
Pitch Rate	12	220	2640
Roll Rate	12	220	2640
Yaw Rate	12	220	2640
Angle of Attack	12	220	2640
Angle of Sideslip	12	220	2640
Control Surface Positions (11)	12	220	29040
C.S. Hinge Moments (11)	12	55	7260
Pitch Angle	12	55	660
Roll Angle	12	55	660
Static Pressure	12	55	660
Total Pressure	12	55	660
Air Temperature	12	55	660
Engine RPM	12	55	660
Engine Fuel Flow Rate	12	55	660
Engine Inlet Temperature	12	55	660
Engine Exhaust Temperature	12	55	660
Discrete Words (6)	12	220	15840
Discrete Words (4)	12	55	2640
			Total = 81.84 kbits/sec

$$= 81.84 \text{ kbits/sec}$$

For expansion capability in the number of data channels, we recommend at least a 100% increase over the baseline requirements. Based on the 39 proportional data channels identified in Table 6.2, the number of proportional channels should be at least 80, including the 120 discrete channels already specified. Since flight control system requirements drove the high sample rate of 220 Hz, the additional 41 proportional channels should be specified at the lower rate of 55 Hz. This increases the required bit rate to

$$\begin{aligned} \text{bit rate} &= 81{,}840 + (41)(55)(12) \\ &= 108.9 \text{ kbits/sec} \end{aligned}$$

An increase in word length from 12 to 16 bits would increase the required data rate by a factor of 4/3, yielding 145.2 kbits/sec. This data rate is still well within the capability of 1 MHz PCM telemetry system.

The uplink to the MURV required 8 12-bit commands and at least 6 12-bit words of discrete parameters. Using the recommended transmission rate of 53.3 Hz (Section 4.3.2), the data rate required for the uplink was calculated to be 8.95 kbits/sec.

6.4 Signal Conditioning

Signal Conditioning refers to the modification of the transducer signals which is done onboard the aircraft and can involve amplification, signal conversion and filtering. In theory, these operations should not affect the information in the signal, but only change it to allow compatibility with the next device in the data system.

Since the electrical signals produced by most transducers are at low voltage and/or power level, it is often necessary to amplify them before they can be transmitted, further processed, displayed or recorded. Common amplifier types include AC, DC, operational and charge amplifiers. The selection of a specific amplifier is based on the transducer output characteristics, the dynamics of the data signal and the desired amplifier output. The process of amplification is sometimes a function of Obviously, the specification of amplification requirements must be done in concert with the selection of transducers and other system components.

As discussed in Appendix O, the outputs of various transducers can be DC or AC analog signals, digital signals, or pulse rates. However, transducer signals must often be in a particular form to be integrated with analog-to-digital converters, telemetry components, and onboard processing. To provide greater flexibility in the choice of transducers, for properties such as range, accuracy, and frequency response, signal converters are provided to convert the transducer signals to the proper form. Analog-to-digital converters commonly required a DC input, varying between 0 to 5 volts, or 0 to 10 volts. Even digital transducer outputs may required conversion to the proper digital format. Additionally, the PCM signals received by the MURV via the telemetry uplink may require digital-to-analog conversion to operate a control process.

Common filtering applications include signal-to-noise ratio improvement, smoothing of data, bandwidth reduction and avoidance of aliasing effects. The filtering that will be required onboard the MURV involves bandwidth reduction and aliasing avoidance. Bandwidth reduction may be required when the transducer selected has a wider bandwidth than can be accomodated by other components, such as the analog-to-digital converter. Also, it is desirable to attenuate signal components that are of no interest to the measurement and which may saturate the transducers or other components. Filtering to reduce aliasing effects is accomplished using low-pass filters and is mandatory for any signals that will be sampled. Since the data measured onboard require sampling in order to transmit them with a PCM telemetry system, this is a necessary design area.

Onboard signal processing is needed when there are multiple sensor sources for one parameter. This situation will require a decision algorithm to establish the voting, averaging, or other possible combinations of multiple sensor inputs. This function must be done onboard the aircraft to provide the backup flight control system with direct data inputs, and to limit the number of telemetered data channels.

6.5 System Flexibility

There were two MURV system objectives defined previously that relate to design flexibility. These were the ability to perform various experimental tests, and the ability to modify the aircraft configuration. Achievement of these system objectives required flexibility in the data acquisition system. The goal in designing the data system was to provide a system that could accomodate aircraft configuration changes and various experimental requirements, with a minimum of modifications. The major factors contributing to this objective were built-in expansion capability in the onboard equipment and flexible software control.

The selection of a PCM telemetry system maximizes the flexibility of the data acquisition system. Subcommutation and supercommutation ensure the most flexibility in data channel assignment, and the variability in word length provides increased accuracy, simply by increasing the word size. Additionally, built-in expansion capability in the number of channels is accomplished by proper selection of analog-to-digital converters, commutators, transmitters and receivers.

Many older telemetry systems rely on hard-wired logic. This makes changes to the system in areas such as sample rate, word size and data channel allocation difficult. The MURV data acquisition system is a software-intensive design, which simplifies making and verifying system modifications. This processing capability was discussed in Section 4.5.4 as one of the functions of the onboard computer system. The use of a standard data bus, such as defined by MIL-STD-1553C, simplifies integration requirements of new components.

6.6 Reliability

Reliable operation of the MURV is dependent, to a large degree, on the reliabilty of the data system. Control of the MURV, especially at beyond-visual-range, relies on the accuracy of the data reception and the reliability of the command uplink. Reliable operation of the data system can be enhanced by component and functional redundancy in the transducers, telemetry links and power sources.

As discussed earlier, primary data recording will be done on the ground. Onboard recording was rejected due to the weight and volume penalties, and because it would not significantly enhance the reliable operation of the MURV. Data reception on the ground is a requirement for MURV operation, therefore onboard recording could provide only a safe-guard against erroneous ground recording or a means for crash investigation. For these reasons, onboard recording was not determined to be a strict requirement, but could be added as the design progresses if the weight penalties were not prohibitive.

VII. Structural Design/Modularity

Introduction Once it had been established in the Conceptual Design Phase that the MURV would be "fighter-like", with a single fuselage (as opposed to one with twin tailbooms, for example) and one or two rear-mounted jet engines, it became possible to begin the structural design of the fuselage, wing, and empennage. While modularity is discussed in each subsystem section in this Volume, the majority of the modularity in terms of aircraft physical reconfiguration was developed in concert with the structural design in this section. And because the wing, all the empennage, the landing gear, and the engine will be mounted to or in the fuselage, much of the modularity of the MURV depends on the design of the fuselage itself. For this reason, the majority of this section deals with the fuselage design and how the other parts such as the wing, engine, and empennage will attach to the fuselage.

It is also important to note that since this design effort was limited to the Preliminary Design Phase, the structural design was pursued only to the point of establishing the layout and rough sizing of the main structural elements. The analysis performed consisted of the application of general structural design concepts and the knowledge of the basic properties of materials. The actual dimensioning, selection of specific materials (aluminum alloys, for example), and prefabricated component selection are left to the Detail Design Phase.

7.1 Objectives

The first step in the structural design was to translate the overall MURV objectives into more specific subobjectives, tailored towards structural design. These "structure subobjectives" were then used to design the structure of the fuselage, wing, and empennage. Since the structure of the MURV would interact closely with other subsystems (such as the propulsion and launch and recovery subsystems), another objective was added: to maximize the structure's compatibility with the other subsystems.

The MURV objective to maximize the number of tests possible led to the structure subobjective to minimize the MURV structural weight. The less the MURV weighed, it was reasoned, the more tests it could perform due to greater performance and the larger allowance for on-board sensors, special control surfaces or other test equipment.

The same logic was used in developing the next two subobjectives: to maximize usable volume and to maximize structural strength. They both would increase the capability of the MURV to perform different tests. Maximizing usable volume referred mostly to the fuselage structural design, although it did come into play in the wing design, when the use of wing fuel bladders was considered. Maximizing strength also helped the MURV objective of minimizing risk, because an aircraft with larger safety margins is less likely to fail in stressful maneuvers.

The objective to maximize the number of tests also created the structure (fuselage) subobjective to allow for different powerplant types. While the Conceptual Design for the baseline MURV specified that it would use one or more jet engines, it was desired that the MURV also be capable of using different engines, such as different kinds of jet, propeller, or turboprop engines. Since the majority of aerodynamic tests could be performed using just the baseline jet engine(s), this objective was not as important as the others.

The MURV objective to maximize the number of aerodynamic configurations created a large number of more specific structure subobjectives. The first of these was to maximize the capability for different wing locations on the fuselage, such as top-, mid-, and bottom-mounted wings, and the ability to move the wing forward and aft on the fuselage. The second was to allow for the maximum number of different wing planforms, such as delta, forward swept, aft swept, and unswept wings. The design was also to allow for wing dihedral and anhedral, as well as to maximize the range of possible root chord lengths.

While the above objectives dealt solely with the wing, the MURV objective to maximize the number of configurations also created the structure objective to maximize the capability for different kinds and locations of empennage. The following kinds of empennage were deemed important: horizontal canards, vertical canards, horizontal tail, vertical tail(s), forebody strakes, and rear ventral fins. For all of these kinds of empennage, the capability of fore/aft movable placement was desired. Also for the horizontal surfaces, it was important that variable vertical placement be possible. And the capability for angled surfaces, i.e. not purely vertical or horizontal, was also desirable. Finally, in the case of the vertical surfaces, the capability for both single or double surfaces was needed.

The two MURV objectives, 1) to maximize versatility in terms of different configurations and 2) to maximize ease of use, more specifically to minimize the amount of test preparation time, combined into the structure subobjective to minimize the amount of work required to reconfigure the aerodynamic surfaces. For example, the structure design should make it as easy as possible to start with the MURV in a canard configuration, remove the canard, add a horizontal tail, and move the wing forward on the fuselage to turn it into a conventional configuration.

The MURV objective to minimize cost applied directly to the structure design. It was broken down into four areas for analysis: materials, labor manhours, amount of off-site work, and the need for custom-built (as opposed to off-the-shelf) components.

The objectives to minimize the amount of test preparation work and to maximize maintainability were very closely related, and for that reason they together created the following structures subobjectives, which applied mostly to the fuselage design: The first was rather specific, to minimize the form drag of the fuselage without external aero shells. If this objective were satisfied to a large enough degree, then the baseline MURV could fly without the need for any external foam shells around the fuselage. This would simplify the amount of test preparation for a large number of different tests. The second fuselage objective was to maximize the accessibility to the contents of the fuselage. This would allow simpler test preparation by, for example, allowing the flight computer to be reprogrammed in place instead of having to remove it. And greater access enhances maintainability by definition. The third objective for the fuselage was to minimise the amount of work required to remove and install the internal equipment. This required simple methods of equipment attachment, but a large part of it also was to try to design the wing attachment so that all of the equipment inside the fuselage could be removed and reinstalled while leaving the wing in place.

The MURV objective to optimize the schedule led to the structure subobjective to keep the design simple so that the time required to build the fuselage would be as short as possible.

Finally, in order to make sure that the structure design work was not performed independently of the other subsystems, another subobjective was added: to maximize the compatibility of the structure with the other MURV subsystems. And since the the degree of interference among the other subystems depended on the fuselage design, this objective also meant that the design of the fuselage should minimize the interference among the other subsystems.

7.2 Fuselage Design

The fuselage design was the logical place to begin the MURV structural design because the fuselage is the central, load-carrying "backbone" of the MURV. The fuselage structure was designed in the most detail (as compared to the wing and empennage) because of modularity considerations: the basic configuration of the fuselage was less likely to change as the MURV aerodynamic configurations changed. Thus, the fuselage needed to be the "common denominator" among all of the possible MURV configurations. It needed to be capable of carrying a variety of wing and empennage combinations, without changing drastically. In that regard, the fuselage design for the twentieth modfication to the MURV would be virtually unchanged from the fuselage design for the baseline. The same could not be said for the wing and empennage. For this reason, the fuselage needed to be designed in a fair amount of detail to insure that it could accomodate a variety of different MURV configurations.

7.2.1 Cross-section Shape The first step in the fuselage design was to establish what the cross-section shape of the main structure would be. The structural concepts ranged between "endoskeleton" and "exoskeleton" types. The endoskeleton types were those in which there was a central structural "spine" to which the equipment, engine(s), fuel tank, and empennage would be attached. Aerodynamic shells would then be mounted around the equipment to give the fuselage good aerodynamic properties. In the exoskeleton concept, the main structure of the fuselage made up a shell, with an open bay for mounting equipment, the fuel tank, and the engine(s). Monocoque and semimonocoque (the way most conventional aircraft are built) designs are exoskeleton type. Exoskeleton designs could be made with enough aerodynamic efficiency so that aero shells would not be required.

Two main types of endoskeleton designs were considered. The first, the I-beam, consisted of a narrow-flange "I"-shaped cross section (Figure 7.1). The wing would be mounted to the top, bottom, or web section of the beam. The other main load-transmitting parts, mainly the empennage, landing gear, and engine(s) would also be attached directly to the beam. The main advantage of this design was that the wing/empennage locations could be changed by bolting them onto the I-beam in different places. It also had two disadvantages: mounting the avionics equipment and fuel tank would be more difficult and it would require aero shells.

The second type of endoskeleton concept consisted of two thin tubular booms mounted side by side and attached to one another at points along the length, much like a ladder. The tubes could have had circular or square cross sections, and possibly could have served as conduits for fuel lines and electrical wires (Figure 7.2). The wing and empennage would have been mounted either directly to the booms or to a structure attached to the booms. This concept had the same disadvantages as the I-beam.

These two concepts were also "turned" ninety degrees and evaluated, but the problems remained. The I-beam, when turned on its side and with its flanges widened, did improve on the

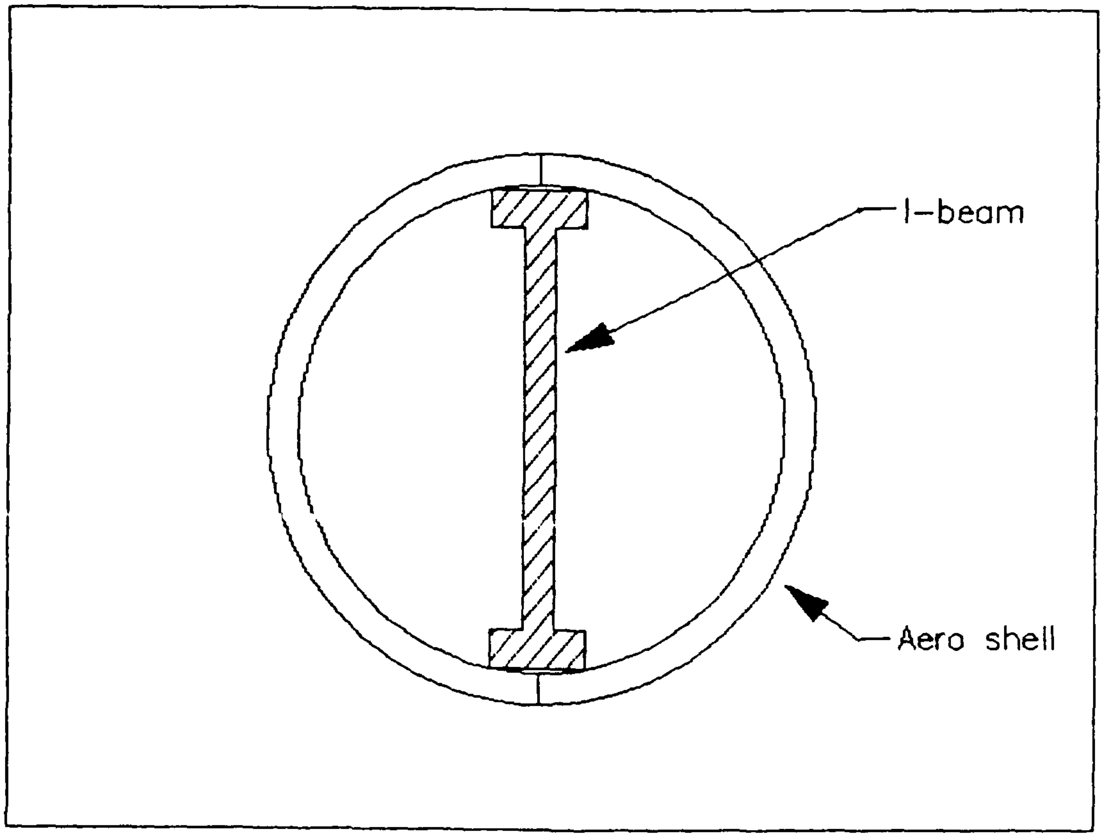


Figure 7.1. I-Beam Fuselage Cross-Section Concept

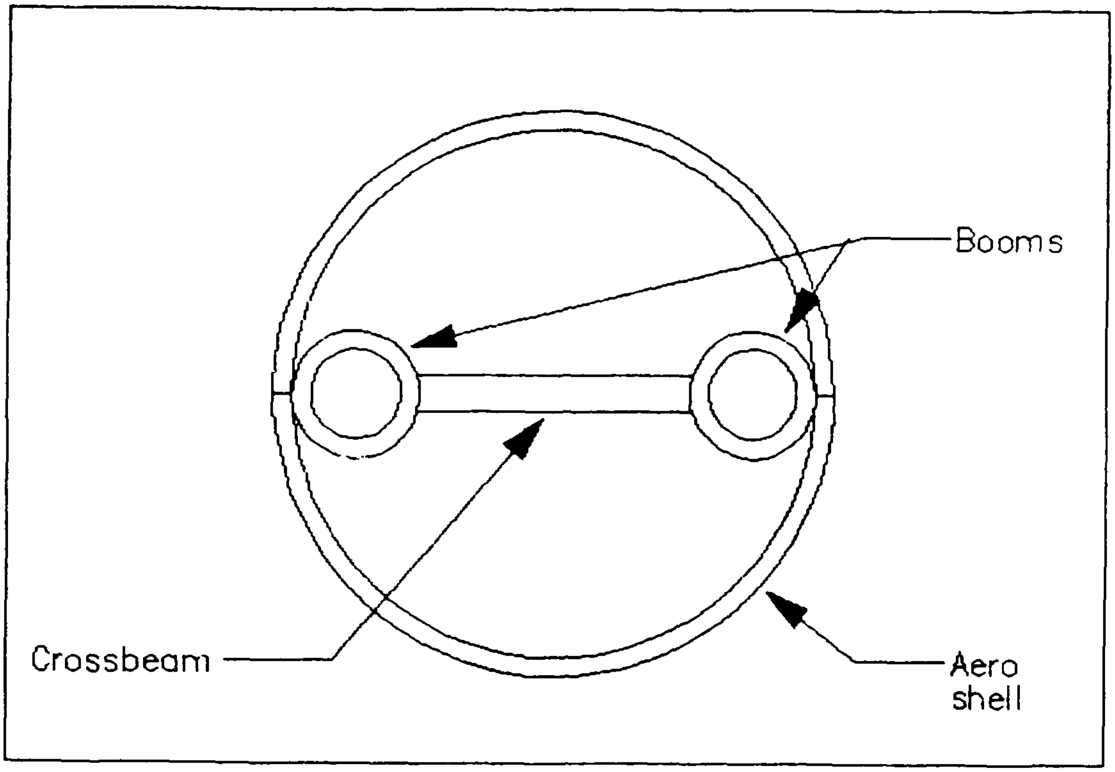


Figure 7.2. Twin-boom Fuselage Cross-Section Concept

problems with the equipment mounting. This concept was called the "H-beam" (Figure 7.3), and with rounded shells covering the open sides, it presented a fairly aerodynamic cross section. Thus, it was in between the endo- and exoskeleton concepts. But it still had problems in that it restricted equipment size, required double access to get to the equipment, and was restricted to a mid-height wing configuration.

There were two main concepts of the exoskeleton type, the circular cross section and the "U-beam" designs. The circular cross section's main problems were that it was more complex to build, access to the interior would have been more difficult, equipment attachment would have been difficult, and attaching the wing to the top or bottom of the fuselage would not have been easy.

The most promising fuselage cross section was the "U-beam" design, in which all of the internal equipment would fit in an internal bay, with a cover over the top (Figure 7.4). The wing/empennage would then bolt to the top or bottom of the fuselage. Whether a mid-height wing

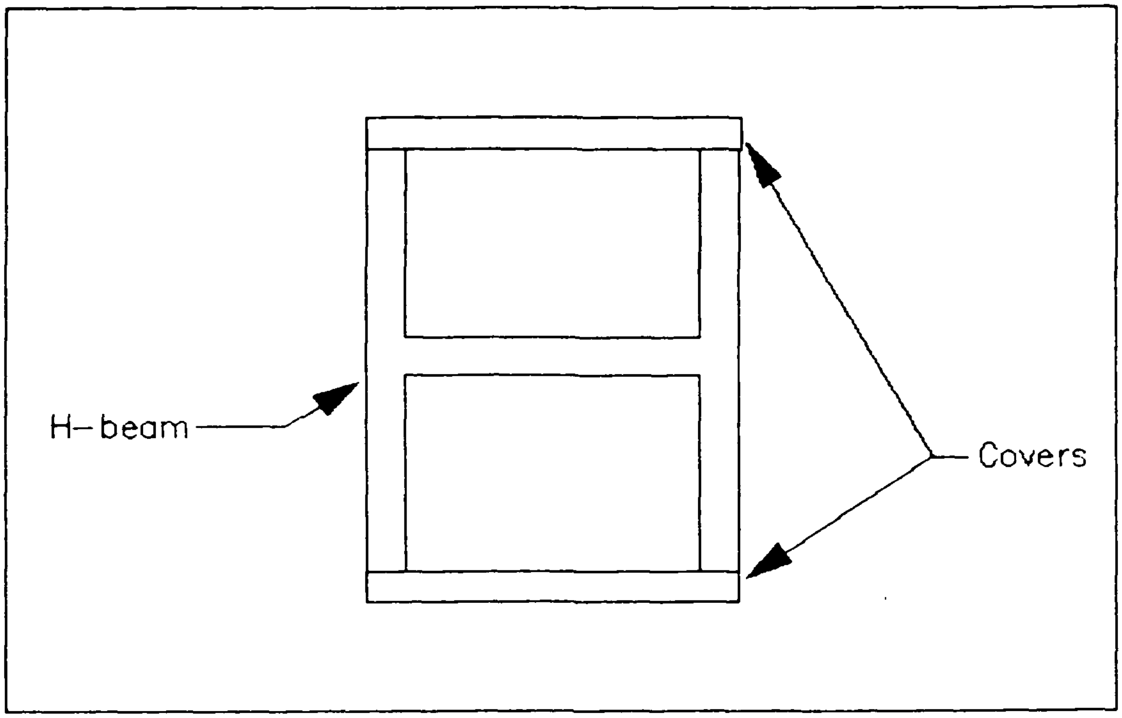


Figure 7.3. H-Beam Fuselage Cross-Section Concept

was possible could not be established at that point. Even though it would make wing attachment more difficult, it was reasoned that it would be best to round the corners of the rectangular cross section in order to make the design aerodynamic enough so that aero shells would not be necessary. So the concept was slightly modified into the "rounded square" design (Figure 7.4).

The rounded square concept had many advantages over the previous cross sections. Even though aero shells were not necessary, they could be added later on the outside of the fuselage if needed for a specific test. Another advantage was that it allowed good access to the interior. Equipment installation in the internal bay was simpler, and the square sidewalls and flat bottom provided a lot of usable volume. With its flat surfaces, it would be easy to manufacture and work on. It was also good from a modularity point of view because the horizontal tails and canards could be mounted at any height. For these reasons, the rounded-square fuselage cross section was chosen. The next step was to determine the optimum internal structural arrangement based on the rounded-square concept.

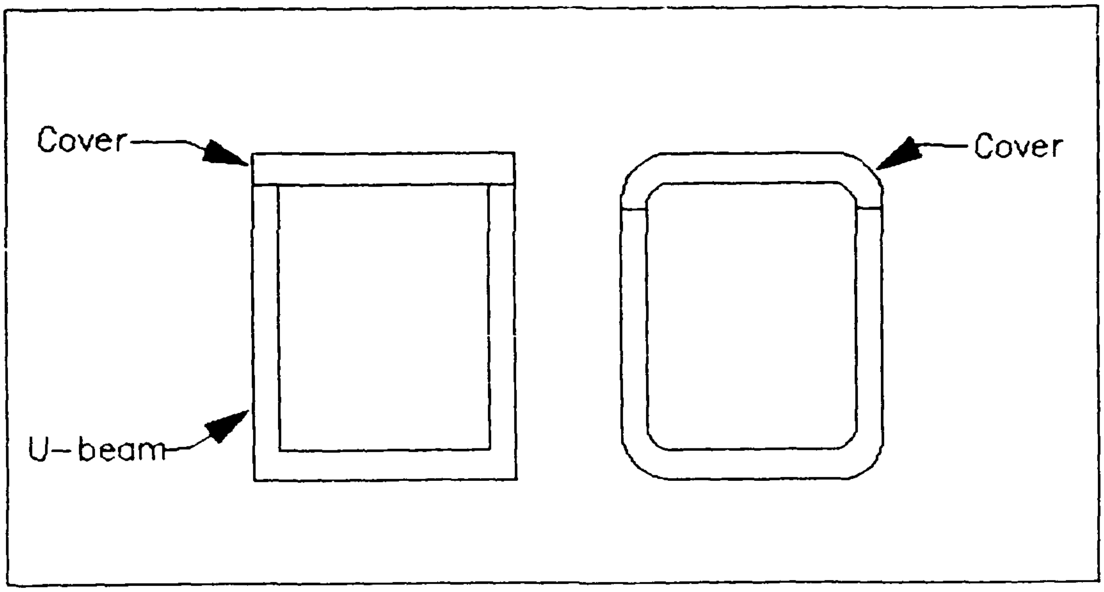


Figure 7.4. U-Beam and Rounded-Square Fuselage Cross- Section Concepts

7.2.1.1 Fuselage Structural Design The first step in structural design was to translate the two-dimensional cross-section into a three-dimensional fuselage shape. Since the wing would be attached directly to the top or bottom of the fuselage, the cross section needed to be constant along the length of the fuselage to allow movement of the wing fore and aft. To minimize drag, the fuselage also needed a nosecone at the front end and tapered "boattail" at the aft end. Thus the fuselage was broken up into three components: the nosecone, main fuselage (constant cross-section), and boattail (Figure 7.5). The main fuselage, wing and empennage designs were all closely related and were developed together, so they will be described in sequence. The nosecone and boattail designs were virtually independent of the wing and empennage, so they will be described after the empennage discussion is completed.

The rounded square design at this point was still very preliminary — no real structural design work had been put into it. If the "U" shape with the cover were adequate in terms of weight, strength, and actual wing mounting techniques, then it could just have been designed in more detail, without any significant changes. This was not the case.

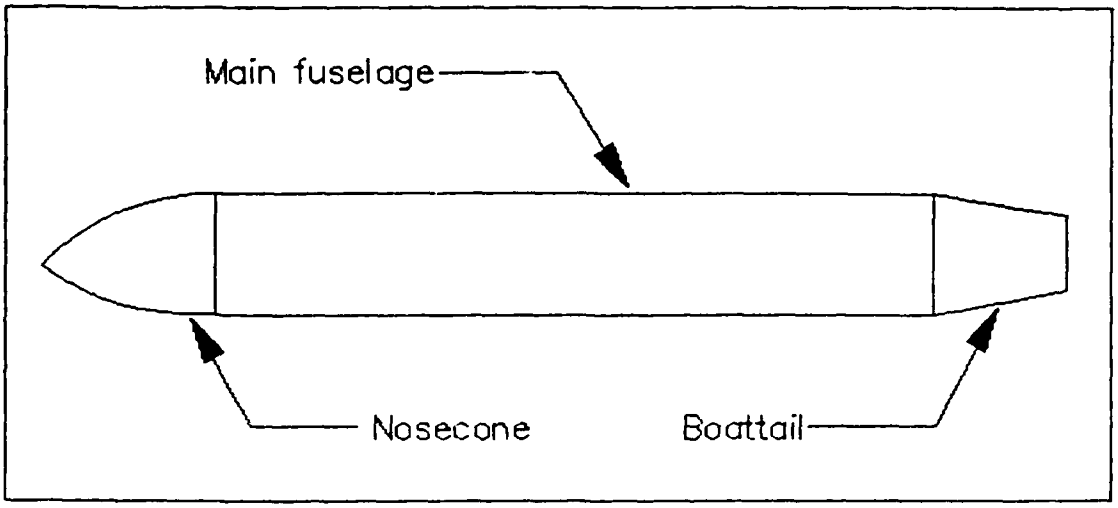


Figure 7.5. Breakdown of Fuselage into Nosecone, Main Fuselage, and Boattail Sections

The two most likely materials for the fuselage construction, aluminum and fiberglass, each had compatibility problems with the rounded square design as it stood. If the entire shell were to be made of aluminum, wings could have been bolted on easily enough, but in order be strong enough it would have been very thick aluminum and hence probably too heavy. A fiberglass fuselage structure would have been lighter, but it would have needed special reinforcement areas where the wings would be attached. In order for the wing to be movable fore and aft, practically the entire fuselage would have required some special wing attachment reinforcement. And in either case, the attachment of the wing to the top of the fuselage would have been weaker because of the joints for the lid. And this would also require removing the wing if access were needed to the interior of the fuselage below the wing. And finally, simply bolting the wing on top or below the fuselage would make an aerodyamically sloppy design—a wing flush with the fuselage would have been much better. So while the rounded square cross section shape was still to be used, the method of acheiving that shape needed work.

A flush wing mounting could have been achieved for the top-mounted wing in the above design if covers were not used in the section of the fuselage where the wing was to be mounted, and instead the wing were mounted directly to the tops of the fuselage sidewalls. Thus, the top of the wing could be flush with the tops of the covers fore and aft of it on the fuselage.

But mounting the wing to the tops of the sidewalls would have been difficult; instead, the wing could be mounted to two "rails" attached to the inside of the sidewalls, at the top of the fuselage. The rails could have tapped holes on their top surfaces every one or two inches along their entire length for wing mounting. The wings would be attached by bolting them to the rails. Box-beams were chosen for the rails for several reasons. The four flat sides were ideal for mounting items to the rails. The outside surfaces would be attached to the fuselage sidewalls, the top surfaces would be used for wing attachment, and the remaining two inside surfaces would be for attaching internal equipment. The hollow interior eliminated the manufacturing complications of drilling and tapping uniform-depth holes. And because all of the material is distributed around the perimeter of the cross section, box-beams are stronger under bending loads than solid beams of equal mass [57:68-69].

If the rails were included for the entire length of the main fuselage, then the wing could be mounted anywhere along its length. The length of the wing root chord would only be limited by the length of the main fuselage. For the wing to be able to move all along the length of the fuselage, the internal equipment could not protrude above the tops of the rails. If the same wing mounting technique were used on the bottom of the fuselage, then the wing could be mounted on the top or bottom of the fuselage, at any point along the length of the fuselage. Thus the design would include four rails, two sidewalls, and removable top and bottom fuselage covers.

The four rails would also provide an excellent framework to which internal equipment could be attached. The equipment could be easily moved fore and aft inside the fuselage. If the sidewalls were made removable as well, then there could be all-around access to the interior of the fuselage. This design was also ideally suited to mounting one or two engines inside the aft end of the fuselage, because the engines could be directly mounted to the main structural elements of the fuselage, the rails.

This process resulted in the box-type structure with four rails, sidewalls, and removable top and bottom covers, as shown in Figure 7.6. The sidewalls would attach to the rails by flush fasteners at two to four inch intervals along the length of the fuselage. The covers would be in approximately one foot long sections, their length depending on the location of external equipment or aerodynamic surfaces mounted to the rails. They would also bolt to the rails at two to four inch intervals.

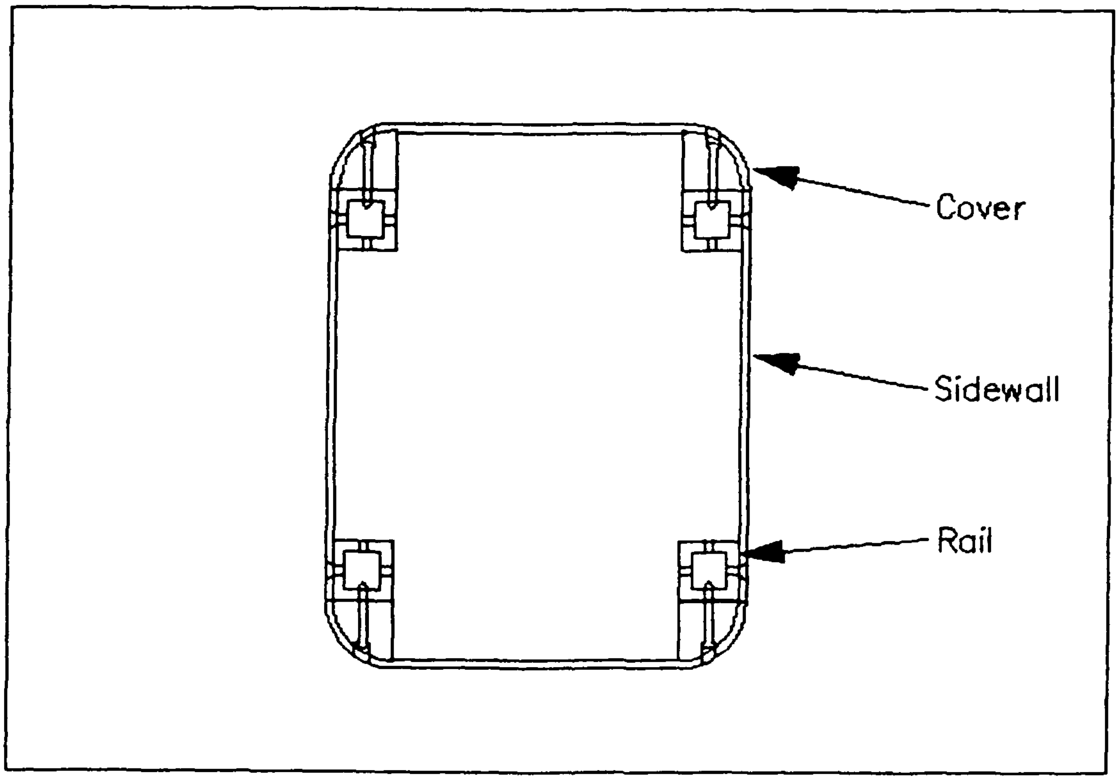


Figure 7.6. Rails/Sidewalls/Covers Fuselage Structure

This design does not require aero shells, but they could be added later if so desired. The flat surfaces of the fuselage exterior allowed easy installation of styrofoam shells.

Aluminum was chosen as the material for the rails, due to its light weight, strength, and capability to be drilled and tapped. The box-beam shape can be easily extruded, so the rails would available and inexpensive. The AFIT Model Fabrication Division has the capability to drill and tap holes in aluminum. The sidewalls and covers would be made of a composite, most likely fiberglass because of its strength, light weight, low cost and the fact that the AFIT Model Fabrication Division has experience working with it [29]. The actual determination of the material to be used would depend on the loads on the sidewalls and covers. Wood was chosen over styrofoam for the cover corner supports because it could better take the bolt loads and the compression loads caused from the vehicle undergoing lateral bending loads.

The fasteners to hold the sidewalls to the rails needed to be lightweight, flush, and easy to assemble and disassemble. The Camloc fasteners mentioned in Volume One, Section 5.6, which are commonly used in fiberglass aircraft construction, are at least one type which meet these requirements [29].

This structural design minimizes weight because all of the elements are load-carrying members. The rails would provide axial strength in the longitudinal direction and would serve as attach points for all load-transmitting components and all internal equipment. The sidewalls would provide the resistance to longitudinal bending and shear. The covers would provide strength for lateral bending and shear. Together, the sidewalls and covers would resist longitudinal twisting of the fuselage. Bulkheads inside the fuselage would provide strength in the vertical and lateral directions. The design also allows for structural stiffeners between the rails, if necessary.

The need to be able to lengthen the fuselage led to the requirement for rail and sidewall joints. The rail joints needed to be strong, and at the same time, not interfere with the mounting of equipment to the rails. These constraints pointed to a rail joint design which would not have any external overlapping plates or sleeves. The best design would fit inside the rails, and yet still allow items to be mounted to the rails in the region of the joint. This led to the design of an internal sleeve, with tapped holes which were the same sise as the fuselage rail holes. In order for a bolt to fit through the rail wall and thread into the inner sleeve without binding, the holes in the rails would have to be larger than the bolts. This allowed rail joints to be bolted together with or without equipment being attached at the joints. Either the bolts would just hold the rails to the sleeve, or they would hold the equipment to the sleeve, with the rail wall in between. This process led to the design described in Volume One, Section 5.6.

The baseline MURV will not have any rail joints because: a) they add weight to the structure, b) they can be added later, and c) the locations of the joints needed for a new configuration will not be known until that new configuration is specified.

There also needed to be joints in the sidewalls to allow them to be lengthened or shortened with the fuselage. They would also allow shorter sections of sidewall, rather than the entire sidewall, to be changed out to accomodate different mid- height horizontal empennage or to access specific areas of the fuselage. The sidewall joints needed to be strong because the sidewalls themselves will be load-carrying members and their structural integrity would be significantly reduced if they had joints which could not transmit the loads. They also needed to be lightweight, and flush on the exterior so as to not increase drag. And finally, they needed to be easy to take apart and reassemble. These design objectives led to the sidewall joint design described in Volume One, Section 5.6.

The fuselage covers would not have structural joints for two reasons. First, the covers will be the primary method of access to the interior of the fuselage, and attaching adjacent covers to one another would make removing covers for access to the fuselage interior more difficult. Second, the rounded shape of the covers, with their wood corner supports, is much more conducive to transmitting loads than the flat sidewalls. Further analysis, taking into account the predicted loads, will need to be performed to verify this design.

7.2.2 Engine Mounting The installation of the engine inside the main fuselage drove some other features of the structural design. The minimum cross-sectional size of the fuselage would be limited by the diameter of the engine at its widest point. The rails would need to be placed around the engine so they do not touch it, and so that the planes made up by the outer surfaces of the rails to not come in contact with the outer shell of the engine. This is illustrated for the baseline configuration in Figure 7.7. The gap around the engine is needed to keep the sidewalls and any hardware mounted on the rails above and below the engine from overheating. An insulation blanket would be attached to the inner surfaces of the sidewalls and to any fuselage covers or hardware above and below the engine. The first estimate of the sise of the gap was approximately one inch. Determining the actual dimension of the gap and thickness and type of insulation requires further analysis.

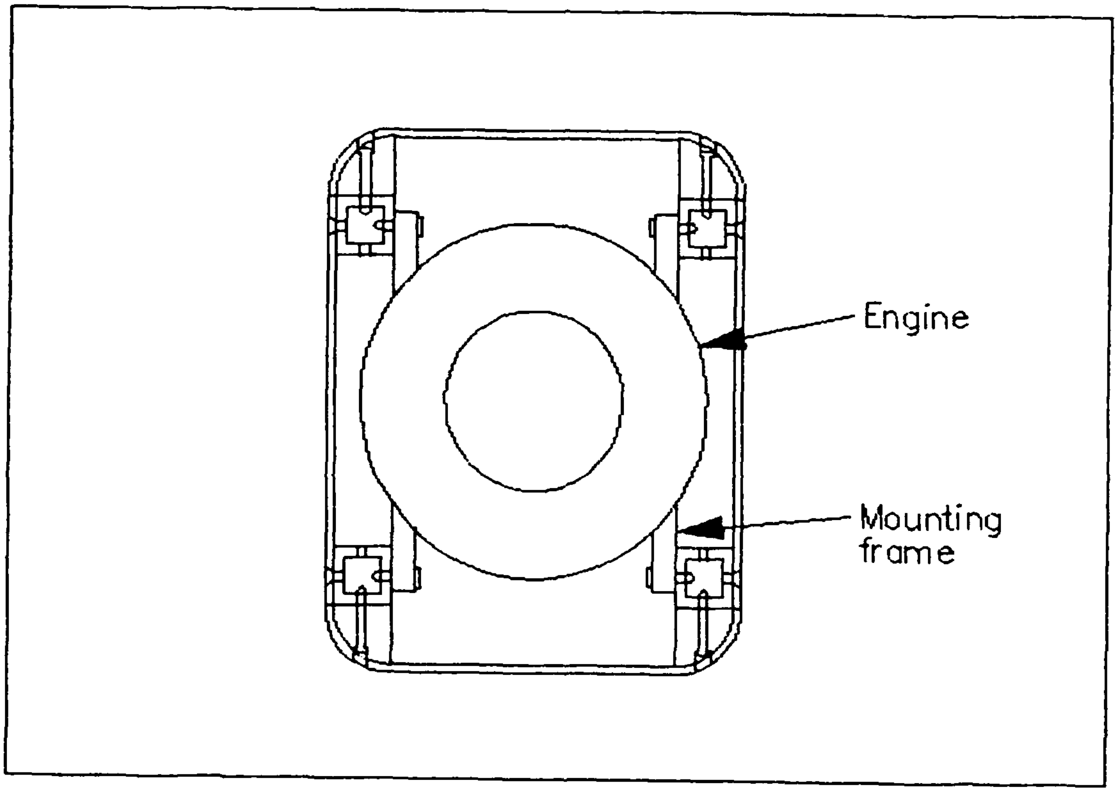


Figure 7.7. Cross-Section View of Engine Mounting in Fuselage

The rail design also allows the intake to pass through the bottom of the fuselage by going between the rails. Because the intake would need to be on the bottom of the aircraft for high angle-of-attack flight, this opened up a problem with wing design if it was to also be on the bottom of the aircraft and of single piece construction.

7.3 Wing Structural Design/Mounting

Before entering into the discussion of the wing design, it is important to note that the aerodynamic design of the baseline wing was already presented in the External Arrangement/Shaping section. This section deals solely with the design of the wing from a structural/fuselage interface point of view, with the special problem of avoiding interference with the intake (for a low wing) as the primary design driver.

In order to maximise the strength of the wing, it needed to be a single unit, without separate left and right wing halves. It needed to have a strong mount to the fuselage rails. This was accomplished by designing the wing with two wing rails, which would mount directly to the fuselage rails with bolts. The wing rails would be as long as needed to distribute the lift loads along the length of the fuselage.

Since the intake could possibly be in the same location as the wing, the wing needed to have an opening in the center for the intake. Conventional model aircraft construction utilises one or two main tubular or I-beam spars which span nearly the entire wing and serve as the main load-transmitting devices to the fuselage. It was decided that the MURV wing would follow the same technique. The area between the wing rails would be open, with the spars being the only structure going across the fuselage opening. With the proper placement of the spars in the wing during its design, the intake could then protrude down between the spars, or completely fore or aft of them. After the installation of the wing and intake was complete, the open area between the rails would be covered by flat composite sheets, mounted to the wing rails.

The wing spars would transmit the loads to the wing rails, which would transmit them to the fuselage rails. The wing rails could be made longer than the root chord of the wing if needed for load distribution. Thus the fuselage would not need a special beefed up section such as the wing box on a conventional aircraft.

This design was also chosen because the open area between the wing rails also allowed for easy access to the interior of the fuselage and the removal and reinstallation of components, without requiring the removal of the wing.

This design also allowed for wing dihedral or anhedral. The wing spars would need to be designed with a bend to allow for this, but the only requirement is that the portions of the spars between the wing rails not interfere with any other hardware. The design also allows for different wing planforms, since the wing mounting technique is independent of planform or root chord.

This concept does not allow for one option, the movable mid-height wing. "Movable" in this context means that it could be mounted at different places along the length of the fuselage, and "mid-height" means any wing height other than top- or bottom-mounted. Any exoskeleton type of design is incompatible with a mid-height wing with carry-through spars because of the interference with the internal equipment and the reduction in strength of the fuselage sidewalls. If the wing were to be movable along the length of the fuselage, then the avionics, fuel tank, and engine/intake would all be affected by the location of the wing spars. Also, the fuselage sidewalls would need to have holes large enough for the wing spars, which would significantly reduce the sidewall strength. Extra bracing could be added, but this would add complexity and weight to the MURV. It would be possible to have a fixed mid-height wing, but it would have to be specially accounted for in a future flight configuration design and would probably require significant modification of the MURV.

While the above discussion refers only to single wings, the fuselage design does allow for dual wings, either one above the other in a biplane configuration or one behind the other in a tandem configuration. For the tandem configuration, the wings would probably not both be on the top or both on the bottom of the fuselage, due to aerodynamic interference. A significant type of tandem wing is the "joined wing" configuration, where a low, forward mounted, aft swept wing and a high, rear mounted, forward swept wing join at the tips. While the baseline MURV will not have this configuration, the fuselage design does allow for it.

The baseline MURV wing will not include fuel bladders. Although they would be beneficial from a fuselage internal volume standpoint, they had many disadvantages. They would complicate the fuel system, opening up more potential failure points. The extra tubing, valves, etc. would add weight. The modularity of the MURV would suffer because wing movement would require rerouting of fuel lines. Each new wing that was made would be more expensive and take longer to make because of the fuel bladders. But most importantly, if wing bladders were required for the MURV to perform a 30-minute flight, then any future wing which could not incorporate fuel bladders (one which was too thin, for example) could not be used.

This does not remove the possibility for a future modification which incorporates wing bladders, if they are needed for a particular test or configuration. But the baseline will not have them.

In addition to the spanwise spar(s) and lack of fuel bladders, the wing will also have the following features: it will be of monocoque construction, with a foam core and a composite skin. This construction technique has many advantages, among them light weight, strength, ease of inspection, and ease of manufacture. Whether basic fiberglass or a more exotic composite will be used will depend on further analysis, specifically what loads will the wing have to take versus cost restrictions.

7.4 Empennage Structural Design/Mounting

In this discussion, the empennage includes all types of fuselage-mounted aerodynamic surfaces. Because they would impart forces on the fuselage, they were designed to have their structural connections to the fuselage rails. For simple attachment and movement, the vertical, horizontal, and angled surfaces and their actuators were designed as self-contained assemblies, which would mount to the top and bottom surfaces of the fuselage rails. To minimize drag, their actuators and electrical connections would be placed inside aerodynamic shells which have the same approximate height as the fuselage covers, and would be shaped to have smooth transitions to the covers fore and/or aft of them. These individual empennage assemblies could be mounted at different locations along the length of the fuselage.

It was also desired to be able to have horisontal empennage at heights partially up the sides of the fuselage. This feature was incorporated in a design in which the empennage would mount to internal frames immediately inside the fuselage sidewalls (on both sides of the fuselage). The frames would attach to one or both of the fuselage rails on that side, and would contain the structural support and the actuators for the control surfaces. The control surfaces themselves would be attached to the frames in cantilever style, with the pivots or other internal control surface structure protruding through holes in the sidewalls.

The holes in the sidewalls would have to be limited in sise and number so that they would not significantly degrade the structural strength of the sidewalls. With composite sidewall construction, the holes would have to be formed into the sidewall sheet as it was being made, to minimise the strength degradation due to the holes. This would require manufacturing new sidewalls each time a control surface was moved.

The empennage located on the fuselage at the same point as the landing gear would have to be designed so as to not interfere with the operation of the gear. This should not be a problem for the front gear, which would only pivot fore and aft, but there may be problems with ventral fins and the rear skids, due to the wide deployment arc the gear would probably have. The problem of interference with the landing gear would also not allow the use of single ventral fins and vertical canards. Single or double vertical tails would still be possible. This was not deemed a great sacrifice since very few aircraft in production today have single ventral fins or vertical canards.

The horisontal tail's vertical placement would be limited by possible internal structural interference with the engine(s). Also, it would not be possible to have the horisontal surfaces at the same height as the rails because the rails would be in the way of the actuation linkage.

The speedbrake could not be mounted under the fuselage because its only possible location would be between the intake and the front gear, and deploying the speedbrake would greatly reduce the airflow to the intake.

The empennage would probably be made of fiberglass, for the same reasons of light weight, strength, ease of construction, and low cost. It is possible that the empennage would be subject to large enough loads to warrant a more expensive composite. That decision is left to the Detail Design phase.

7.4.1 Summary of Fuselage Main Design Features The following is a summary of the main features of the structural design of the main fuselage, wing, and empennage:

- It allows the wing to be flush-mounted on the top or bottom of the fuselage.
- It allows a one-piece wing with carry-through main spars.
- It eliminates the need for a special reinforced section of the fuselage for wing mounting.
- It allows virtually unlimited empennage arrangements.
- The internal equipment could be easily mounted, and could be moved fore and aft inside the fuselage as needed.
- It allows for virtual unrestricted access to the interior of the fuselage regardless of wing location.
- It allows the engine(s) to be directly mounted to the rails, the main structural elements of the MURV.
- It allows the wing to be in the same area of the fuselage underside as the intake, without interference.

7.5 Nosecone/Boattail Development

Once it was decided that the fuselage would have a constant cross-section for modularity, there became a need for a tapered nosecone and rear boattail to minimise drag. The objectives for the nosecone and boattail designs were the same as for the main fuselage. The most important of these were to minimise form drag, minimise weight, maximise structural strength, maximise usable volume, maximise the potential for reconfiguration, and to maximise the access to the interior.

7.5..1 Nosecone Development To minimise drag, the nosecone was designed to be flush with the outer surface of the fuselage (with covers) at its attach point, and taper gradually to a rounded or pointed tip. All of its fasteners, attaching it to the fuselage, were put inside the fuselage so that the nosecone would have a smooth aerodynamic shape, thus reducing drag. In order to maximise access to the contents of the nosecone, it was designed with hinge mounts so that it could swing horizontally about a pivot at one side of the nosecone/fuselage joint, exposing the contents of the nosecone and forward fuselage.

The nosecone was chosen to be made out of a composite material, to take advantage of composite materials' strength, light weight, and ability to be molded into smooth contours easily. Fiberglass is the most likely candidate because it is the least expensive and the relatively small loads on the nosecone should not require a stronger composite. Further analysis needs to be performed to verify this. The composite would be molded around a wood framework to give it structural strength and serve as a mounting framework for internal equipment. The nosecone would also need a window or opening for the camera to look through.

7.5.1 Boattail Development The "boattail" of the MURV refers to the end section where the fuselage tapers from the rounded square cross section to a smaller circular cross section around the end of the engine jetpipe. The purpose of the boattail is to reduce form drag by keeping the air flow attached to rear of the fuselage in smooth streams and thus eliminating as much of the stagnated air behind the fuselage as possible.

The boattail was designed to be easily removable for modularity and to allow access to the rear of the engine. It was not feasible to have it pivot like the nosecone because the jetpipe would be in the way. The design to make it removable allows the boattail to be removed and replaced with a different design. For example, even though the baseline MURV will not have thrust vectoring, the boattail could be replaced with one which is structural in design and contains the necessary paddles and actuators to provide thrust vectoring, without changing the main fuselage. The removable boattail design also allows simple engine removal by first detaching the intake from the front of the engine, removing the boattail, removing the engine fasteners to the rails, and sliding the engine(s) and support frame out the back of the fuselage.

The boattail was designed to be made of a composite material, probably fiberglass, for the same reasons as the nosecone. It would probably need to be a high-temperature type composite, possibly with insulation blankets glued to the inside surface. Further analysis needs to be performed to determine the actual temperatures the boattail structure may encounter, in order to choose the materials and insulation scheme. It also needed a inner structure made of wood or a higher temperature compatible material, to provide strength for the jetpipe vibration supression function it would perform, and to serve as a mounting framework for the speedbrake mechanisms.

7.6 Electrical Modularity

The electrical modularity of the MURV discussed in this section deals with the mechanical side of the electrical systems: the design of wires, connectors, and how they are arranged inside the MURV. The electrical system of the MURV consists of two areas: power supply/distribution and signal distribution. The power supply will be modular in that additional batteries can easily be added to the main and backup power sources if necessary. Whether this will be accomplished with plug-in type batteries or with ones that require wiring into the existing system will depend on the detailed design. The modularity of the signal distribution system is more complicated.

The "signal distribution system" refers to the wires running between the onboard avionics computers and the actuators and sensors. Since the actuators and sensors will vary both in number and location, their connections to the avionics computers needed be easy to disconnect, reroute, and reconnect. This was accomplished by using wires with metal contacts, which would fit into plastic or metal connectors. The connectors would range from single wire capacity up to perhaps thirty or forty wire capacity. The connectors would allow bundles of wires to be simultaneously disconnected from each other or from components.

The individual wires were designed to be detachable from the connectors, so that a single wire could be replaced without removing and rewiring an entire harness. It also allowed all of the wires attached to a connector to be separated from it, so that individual connectors could be replaced. This would allow the most efficient use of wires and connectors to be attained for any MURV configuration. The connectors were also designed to be easily separated to make components with many wire attachments easy to remove.

The bundles of wires extending down the length of the fuselage would not be covered in sheathing, to allow for easy removal of individual wires. But they would be routed through wire bundle holders, stationed at intermittent points along the length of the fuselage, to keep the wires organised and out of the way.

Since the wings would be made with all their control surface actuators inside the wing airfoils, they would need to be prewired, with wires with connectors protuding from each wing rail. Any other component with inaccessible actuators would also need to be prewired in the same way, with the connectors inside the fuselage cavity when the component is installed. This is to insure that there would be no wires on the exterior of the MURV, thereby minimizing drag.

The capability to remove individual wires and replace connectors allows for virtually unlimited flexibility of sensor, actuator, and avionics computer placement, with a minimum of work required to reconfigure the wiring system.

7.7 *Design Evaluation*

The fo. ·ing is a summary of of how well the structural design of the fuselage, wing, and empennage met the structure subobjectives. By satisfying the structure subobjectives optimally, it also implicitly satisfied the overall MURV objectives optimally.

It minimized weight and maximized structural strength by using lightweight, high strength materials and by having every part of the fuselage structure be a load-carrying member. It maximised usable volume in two ways. First, the flat interior surfaces allowed the box-shaped computers, batteries, and sensors to stack and fit snugly inside the fuselage without wasting space. Second, the constant cross-section allowed interior equipment to be moved fore and aft without having to be rearranged. This allowed great flexibility in the equipment stowage, making the existing volume more usable. The option to use the interior of the wing to store fuel was deemed not worth the added problems.

The objective to allow the fuselage to accomodate *different engine types* was partially satisfied. Different models of jet engines and propeller engines could be installed, the only restriction being that they must fit into the cross-section of the fuselage.

A sacrifice had to be made in meeting the objective of maximizing the capability for different wing locations. The capability to have a movable mid-height wing was not realized. But the design does allow for the full fore and aft movement along the fuselage of both top- and bottom-mounted wings.

The next three objectives were all satisfied to their fullest extent. They were to maximize the number of different wing planforms possible, to allow for both wing anhedral and dihedral, and to maximize the range of possible wing root chord lengths.

The objective to maximize the capability for different kinds of empennage was also satisfied to a great extent. The fuselage design allowed for horizontal, vertical, or angled canards or tails, and also forebody strakes. It also has the capability for a T-tail or a fuselage-mounted speedbrake.

All of these types of empennage could be moved fore or aft, and the horizontal surfaces could be moved up or down on the fuselage sidewalls. There were a few restrictions on the empennage. The horizontal control surfaces could not be at the same height as the fuselage rails, and the placement of control surfaces on the nosecone or boattail would require modification of those structures. Also, single vertical surfaces in the regions of the front and rear landing gear were not possible.

The next objective, to minimize the amount of work to reconfigure the aerodynamic surfaces, was well satisfied by the structure design. For the wing and top- and bottom-mounted empennage, the procedure would be to unbolt them from the rails, relocate them, and bolt them back down. The affected top and bottom fuselage covers would have to be rearranged or replaced. No internal equipment would have to be moved because it would not protrude into the areas above the top surfaces of the top rails and below the bottom surfaces of the bottom rails. The procedure for the mid-height horizontal empennage would be more difficult because it would require replacing the affected sidewalls with new sidewalls premade with holes for the pivots or actuator linkage.

The cost was minimized by this design in several ways. First, readily available materials and components were chosen. Second, the construction work required was kept simple enough so that it could all be done by the AFIT Model Fabrication Division, with a minimium of labor manhours. This also helped satisfy the objective to minimize the time required to build the fuselage.

The rounded-square cross-section design, along with the nosecone and boattail, also met the objective of minimizing the form drag without the use of aero shells. Aero shells would not be needed on the baseline MURV for it to be able to fly.

The objective to maximize the accessibility to the contents of the fuselage was also well satisfied by this design. In the sections of the fuselage where there was no wing or empennage attached, the removable sidewalls and covers allowed complete access to the fuselage interior. The empennage might restrict access somewhat if left in place, but the wing design, with its open area between its rails, still allowed total access to the contents inside the fuselage. And the intake, where it extended below the fuselage, restricted access from that direction. The pivoting nosecone and removable boattail also allowed excellent access.

This structural design also did a good job of minimizing the amount of work required to remove equipment from the fuselage. The scheme of mounting equipment to the rails allowed for simple removal and installation. The open area in the wing made it possible to remove equipment "through" the wing without having to remove it. Removing equipment from the main fuselage would only involve removing the appropriate covers or sidewalls and removing the component. Nosecone-mounted equipment would be removed either by removing one of the covers or sidewalls adjacent to the nosecone or by pivoting the nosecone out to remove the component. Removal of the engine would require first removing the boattail.

The structural design and the integration of subsystems into the fuselage minimized the amount of interference among the subsystems. The most notable conflicts were between the rear landing gear and any aft ventral fins, the engine and any mid-height horizontal tail, and the wheelwell for the forward gear and the onboard computers.

Overall, this MURV fuselage, wing, and empennage structure design allows great flexibility in aerodynamic configurations with a minimum of weight and cost.

VIII. Remote Cockpit Development

8.1 Definition of Scope

The first step in establishing the design of the remote cockpit was to determine its boundaries within the MURV ground system. By its very nature, it was closely integrated with the ground portions of the data acquisition and flight control systems. It was established that the cockpit would not be involved in any real-time data readout for engineering analysis; the data acquisition system would perform that task. The only data the remote cockpit would display would be that required for the pilot to fly the MURV safely and perform the flight tests. The data acquisition system would provide all of the necessary data to the cockpit, in a form that the cockpit instruments could use directly.

The pilot's commands would be input to the flight control system via the cockpit controls, and the flight control system would translate those commands into actuator commands. The actuator commands would then be sent to the vehicle by the telemetry uplink, as described in the sections discussing the data acquisition system. Actuator commands which will not be under the control of the flight control system, such as landing gear deploy commands, will bypass the flight control system and go directly to the vehicle via the telemetry uplink. Thus, the cockpit will be a computationally "dumb" system, its only job to receive the data from the data acquisition system and display it to the pilot, and to take the pilot's control inputs and relay them to the flight control system (or directly to the telemetry uplink, in some cases).

8.2 Objectives

The remote cockpit was designed by using objectives specific to the remote cockpit, which were derived from the overall MURV objectives described in Volume One, Chapter II. Only those MURV objectives which applied to the preliminary phase of the cockpit design were used. The cockpit objectives are explained below.

The system objective to maximize versatility in terms of the kinds of different tests which could be performed created the cockpit objective to maximise the capability of the cockpit to allow different tests to be performed successfully. This meant that all the necessary controls and information for as many different tests as possible be available to the pilot.

The MURV objective to maximise the versatility in terms of aerodynamic configurations led to the cockpit objective to maximise the capability of the remote cockpit to control various MURV aerodynamic configurations.

The objective to minimize the cost of the MURV translated directly into the objective to minimize the cost of the remote cockpit.

The MURV objective to maximise ease of use created several objectives for the cockpit. The first was to maximize the mobility of the cockpit, i.e. to make it as easy as possible to transport to the test site and set up. The second was to minimise the electrical power requirement of the cockpit, which came from the possibility that the ground station may be generator powered, and it should be able to get by with the smallest generator possible. Third, the cockpit controls and instruments should be as similar as possible in design and function to those for conventional remotely piloted aircraft, so that a person with previous experience in flying remotely piloted aircraft (including radio control models) would need a minimum of training (or retraining) to fly the MURV.

The objective to maximise ease of use, combined with the one to minimise risk, created the cockpit objective to minimize pilot workload. Minimizing pilot workload decreased the risk by allowing the pilot to concentrate on his job, to fly the MURV, without being distracted with extraneous tasks. It maximized ease of use by reducing the amount of training required for the pilot.

The MURV objective to minimise risk also created the cockpit objective to minimise risk. The results of the conceptual design phase indicated that the MURV would cruise at approximately 0.4 Mach, and traveling at this speed in a straight line, it would quickly be beyond visual range.

Thus a major factor in minimising risk was to be able to know the location of the MURV at all times during flight.

Finally, since the remote cockpit would interface closely with the data acquisition and flight control systems, an objective was established to maximise the compatibility with these systems.

8.3 Remote Cockpit Design Development

8.3.1 General Configuration The first step in designing the cockpit was to determine its general configuration. The first decision was whether to make it inside a vehicle or outside. For the times when the MURV would be cruising beyond visual range, whether the cockpit was inside or outside had no effect on how well the pilot could fly the MURV. But the case when the MURV was flying low in the immediate vicinity of the runway area (such as immediately after takeoff or during preparation for landing approach), the pilot would need to see the MURV directly due to the added risk of an accident (due to low altitude flight) and harm to personnel or equipment. And since the MURV could fly behind the pilot, he would need to be able to move around to keep the MURV in view, and still remain close enough to the cockpit instruments to be able to read them quickly.

Since no vehicle of reasonable cost would have the visibility and room to allow this, an outside cockpit would be needed. The need to be able to move around adds another requirement to the design, either that the entire cockpit be on a rotating pedestal or that the most critical controls be on a mini-control box which could be removed from the main cockpit control panel and held by the pilot as he follows the MURV visually. The latter would be less expensive and lighter for transport to the test site. It would have the necessary control functions on a handheld box, much like a model aircraft radio control box, connected to the main cockpit with a cable.

The handheld box could either duplicate the most critical controls on the main cockpit, or it could fit into a recess on the main cockpit console and its controls would also be the main controls for the cockpit. Each concept has its disadvantages. If the controls were duplicated, it might cost more and there would have to be a method for determining which sets of controls would have the control authority for different phases of flight. Mounting the handheld box into the cockpit console might require a special mounting mechanism and attention to the human factors involved in placing the controls. But either approach seems feasible at this point.

8.3.2 Identification of Controls and Instruments The next step in designing the cockpit was to establish what controls and instruments would be needed to allow the pilot to fly the MURV and conduct the various flight tests safely. In this context, *instrument* refers to any kind of method of getting visual information to the pilot, whether it be a video display, an LED, or an actual cockpit instrument such as an altimeter.

First, the controls and instruments were broken down into three categories, *Universal*, *Configuration Dependent*, and *Test Dependent*. Those items in the *Universal* category were needed to control the MURV regardless of its physical or software configuration or the test it was performing. An example of this would be an airspeed indicator. *Configuration Dependent* controls and instruments were those which may or may not be needed, depending on the configuration of the MURV. For example, a thrust vectoring control would not be needed for the baseline configuration, but might be needed for a later configuration. The items in the *Test Dependent* category were those which were needed only for a specific test or series of tests.

8.3.2.1 Universal Controls The *Universal* controls were further divided into two categories, *Required* and *Unnecessary*. *Required* controls were those which were deemed mandatory to fly the MURV. These were constraints on the cockpit design. Other controls were determined to be *Unnecessary* because they either conflicted with the cockpit design objectives or they increased pilot workload at critical times without a corresponding increase in MURV performance.

Required Controls

- Pitch/roll control (stick)
- Throttle/rudder control
- Front gear deploy
- Rear gear deploy
- Nosewheel steering engage/disengage
- Backup flight control system engage/disengage
- Initiate self-test

It was decided to have the pitch/roll control be a two-degree-of-freedom joystick, sprung to return to center, to make the MURV controls common with conventional aircraft and radio control model aircraft controls.

Before going any further in the discussion of the stick design, *rate* and *position* commands will be explained. An example of a *rate* command is a conventional aircraft roll control. Control stick (or yoke) roll inputs are rate commands: a right stick command would cause the aircraft to roll continuously until the command is removed. Thus the position of the stick controls the *rate* at which the aircraft rolls. An aircraft pitch command is an example of a *position* command. Pulling back on the stick would cause the aircraft to pitch up to a certain angle (position), and stay there.

The MURV control stick needed to have the qualities of a conventional elevator/aileron control, in which the pitch command is a position command and the roll command is a rate command, to maximize commonality with existing aircraft controls. At the same time it also needed to be able to provide more general tyoes of commands, such as a roll position command, for supermaneuverability. Both of these would be accomplished by having the raw stick position information sent to the flight control system, which would use its control laws to transform the stick position into control surface actuation commands. The flight control system would determine, in how it converts the stick information into actuator commands, how the stick position would affect the behavior of the vehicle.

The throttle and rudder commands were placed on the same two-degree-of-freedom stick to make the MURV controls similar to radio control aircraft controls. For the same reason, the rudder control would return to center, and the throttle control would stay in position. This would also make them more like conventional aircraft controls in general.

The front and rear gear deploy controls would cause their respective landing gear to deploy. There is no retract option because the landing gear will be one-time deploy type (see Volume One, Section 5.4). The nosewheel steering switch was needed for crosswind takeoffs and landings, at low speeds where the rudders are not effective. It would convert left/right rudder commands into left/right turning commands for the nosewheel. For takeoffs, it would steer the dolly nosewheel, and during landings, the MURV nosewheel.

The backup flight control system engage/disengage switch would allow the pilot to command the backup flight control system to take over control of the MURV and enter into its preprogrammed maneuvers. This was needed, for example, if the MURV was flying beyond visual range and the downlink telemetry stream was lost. Since the pilot would know nothing about the attitude or speed of the vehicle, he would give control to the onboard backup flight control system, which would put the MURV into a series of maneuvers. The last maneuver, a slow descending spiral, would eventually result in a controlled crash landing if contact were never regained. If contact were reestablished, the pilot could disengage the backup flight control system and continue the flight.

The switch to initiate a self test was needed for preflight checks. It would insure the integrity of both the primary and backup portions of the flight control system, before the MURV left the ground.

Unnecessary Controls

- Rudder pedals
- Direct variable controls of other control surfaces

Rudder pedals were discarded in favor of a stick-type rudder control for two reasons. First, conventional radio controlled aircraft use a rudder control on the throttle joystick, and one of the objectives of the design was to maximize the commonality with current radio- or remote-control aircraft control schemes. Second, for the times when the MURV would be flying around the runway area, the pilot would need to get up from his seat at the cockpit and move around, so that he could follow the MURV visually during its low flight. Since he would not be able to control rudder pedals standing up, the rudder needed to be hand-controlled.

It was felt that giving the pilot control of individual sets of continuously variable control surfaces, such as ailerons or canards, would greatly increase the pilot workload, especially during critical maneuvers. These control surfaces would all be controlled by the flight control computer. The control of discrete position types of control surfaces such as flaps would not overtax the pilot, and they are addressed below.

8.3.2.2 Configuration Dependent and Test Dependent Controls These controls would be "programmable" in that they could be used to perform different functions depending on the configuration of the MURV or the test it was performing.

- Flight mode select—approximately 6 flight modes
- Preprogrammed maneuver engage—approximately 4 maneuvers
- 10 Return-to-neutral switches

  Examples:

  - Strakes
  - Speedbrake(s)

- Spoilers
- Flaps
- Leading-edge slats
- Engine igniter energize
- Outrigger wheel controls
- Main chute deploy
- Spin chute deploy
- Drogue chute deploy

The first item, the flight mode select, was needed to allow the pilot to switch among the available flight modes in the flight control computer. The flight modes available will depend on the configuration of the MURV or the test it is performing. The capability to select from different flight modes was needed for the MURV to be able to test different supermaneuverable flight regimes. The number of modes which could be selected came from an estimate of the maximum number of flight modes that the flight control system could contain at any one time. This number will probably change as the design of the flight control system is further defined.

The "conventional" mode, in which the MURV would behave like a traditional three-axis controlled aircraft, would always be available. This is so that the MURV would always have a method of control common with other remotely piloted aircraft, to make it easier to fly by a pilot with radio control or other aircraft experience. Also, its allowance for direct rudder control was needed for crosswind takeoffs and landings.

The preprogrammed maneuver engage was needed to allow the pilot to initiate a specific series of maneuvers for testing purposes. By having the maneuvers stored in memory and executed by the flight control system, a significant improvement in repeatability could be obtained as compared to piloted flight. The number of maneuvers was also an estimate of the maximum available at any one time in the flight control computer.

The return-to-neutral switches were needed to control discrete position control surfaces or other MURV hardware. The number of switches needed is a first estimate, based on the projected maximum number of separate items which could realistically be needed for a MURV configuration. This number is very rough, and will probably change as the MURV design becomes more defined. "Discrete position control surfaces" are those which have two or more positions, but cannot be set to any positions in between. For example, a flap could have four positions: fully retracted, extended 10 degrees, extended 20 degrees, and fully extended. The switches could be used for those items over which the pilot has complete control, or they could also change the positions of surfaces which can also be controlled by the flight computer. The switches were designed to return to a neutral position so to not indicate the status of the device by their position. This is to eliminate confusion if, for example, the pilot were to extend the speedbrake and the computer were to retract it later, there would be no disagreement between the switch position (extended) and the control surface position (retracted) because the switch would in fact not indicate any position. All position information would be displayed by the indicators described later in this section.

The number of the test- and configuration-specific controls was limited to keep the cost of the remote cockpit, flight control system, and data acquisition system down. Also, it was important not to overtax the pilot with multitudes of switches. It limited the number of control surfaces or other devices which could be directly controlled by the pilot, but it would take an extremely complex MURV configuration to exceed the capability of 20 controls.

8.3.2.3 Handheld Box Controls

- Pitch/roll control (stick)
- Throttle/rudder control

- Front gear retract/deploy
- Rear gear deploy
- Nosewheel steering engage/disengage
- 4 Return-to-neutral switches

The handheld box would be used during times when the MURV is in close (approximately 1/2 mile) proximity to the runway, the critical portions of which would be takeoff and landing. The "conventional" flight mode of the MURV would be the mode most likely used during takeoffs and landings because the uncoupled rudder control allows better control during crosswind takeoffs or landings. The "conventional" flight mode required both stick and throttle/rudder controls.

The landing gear retract/deploy control was needed for takeoff and landing, and the nosewheel steering engage/disengage was needed for taxiing and crosswind takeoffs and landings.

Four return-to-neutral switches (as part of the ten total), were needed for such things as speed brake extend, drag chute deploy, or wingtip gear deploy, depending on the different MURV configurations. This number was also derived from an estimate of the maximum number of items which would be need to be controlled during flight near the runway area. The number was limited so as to keep the handheld box from becoming too large and unwieldy.

8.3.2.4 Universal Instruments The *Universal* instruments were also divided into 'wo categories, *Required* and *Desired*. The *Required* instruments were necessary to fly the MURV safely, and were constraints on the design. The ones in the *Desired* category were considered "nice to have", but not absolutely necessary. It was felt that having extra instruments which were not necessary did not add significantly to pilot workload; they were there for the pilot to look at or ignore as he desired.

Required Instruments The two main objectives of the remote cockpit were to allow the pilot to fly the MURV with a minimum of risk, and to allow him to fly it with an accuracy that allows the gathering of flight test data. These objectives helped establish which instruments would definitely be needed in the remote cockpit. These instruments are listed below:

- Attitude Direction Indicator (Artificial Horizon)
- Airspeed Indicator
- Vertical Velocity Indicator
- Altimeter
- Angle of Attack
- Health indicators (with audible alarm)
  - Uplink status
  - Downlink status
  - Flight control computer status
  - Backup flight control system engaged status
  - Backup flight control system health status
  - Main onboard electrical power status
- Front gear position
- Rear gear position
- Video
- MURV location display

The first five instruments above are the basic instruments found on almost any aircraft today. They were deemed necessary for the pilot to be able to safely fly the MURV. The attitude direction indicator would require roll and pitch postion information from the data acquisition system. The airspeed indicator would require airspeed calculated from the static and total pressure by the data system. The data acquistion system would also calculate the vertical velocity using the change in static pressure, and supply it to the vertical velocity indicator. The altimeter would require altitude data, calculated from static pressure by the data acquisition system. The six most critical health indicators were also required to keep the pilot informed of the health of the MURV during flight, so he could perform the necessary recovery or abort procedures. They were tied in with an audible alarm so that the pilot would be sure to notice a problem immediately.

The gear position indicators were included to let the pilot know what the actual gear positions were, as opposed to the commanded positions. These positions would be determined by sensors on the aircraft and continuously telemetered down. These were needed in case one of the spring-powered landing gear deployed accidentally in flight.

The video display would be needed mostly to allow the pilot to line up the MURV with the runway during its approach and landing. It would also be helpful in giving the pilot and other observers a visual reference in order to "feel" what the MURV is doing during flight beyond visual range. A black-and-white display would be adequate and would be used to keep the cost down.

The MURV position display was needed for safety reasons. Since the MURV would fly beyond visual range, the pilot would need to know its location at all times so that he would not fly beyond the designated test area and so that he could navigate his way back to the runway for landing. The display would also be useful for establishing the location of the impact if the MURV were to crash. The display would consist of an x-y grid with the ground station and other landmarks plotted on it. It would display the location of the MURV on the grid as a moving point or cursor. It would be desirable for the display to also give the MURV's heading (direction of flight), to help the pilot guide the MURV back to the runway. If this is not possible, the heading will have to be included as another instrument on the cockpit console. The tracking system itself is discussed separately in Section X.

Desired Instruments These instruments would aid the pilot in flying the MURV, but they were not considered mandatory. Having these instruments in the cockpit would not add any requirements for additional sensors on board the MURV, since all the necessary information would already be gathered by the data acquisition system. The only additional work required would be to compute the data into the appropriate parameters, and put them in a form compatible with the cockpit instruments.

- Sideslip angle
- Turn/slip indicator
- Machmeter
- G-meter
- Fuel level (including fuel flowmeter status)

The sideslip angle would be supplied by the data acquisition system from an on-board sideslip sensor. The turn/slip indicator would require bank angle, normal acceleration, and lateral acceleration as inputs. The data acquisition system would calculate the Mach number using static pressure, total pressure, and air temperature and supply it to the Machmeter. The g-meter would use normal acceleration. Since the pilot would not physically be in the MURV, the g-meter would be useful to tell him the g forces that the MURV was undergoing. It would not be absolutely necessary because the flight control computer would be programmed to limit the maximum g-levels so that the pilot would not overstress the airframe. The fuel level indicator would not be an actual readout of the fuel in the tank; that was considered to be too difficult to obtain. Instead, the data acquisition system would use the initial full level and the fuel flow rate to calculate the approximate amount of fuel remaining. If a fuel level indicator is included in the design, then an indicator showing the status of the fuel flowmeter. This is to let the pilot know if the fuel flowmeter has failed and to disregard the fuel level readout.

8.3.2.5 Configuration-Specific and Test-Specific Instruments While it was impossible to know exactly which instruments would be needed for all the possible configuration variations or different tests, an attempt was made to guess as many as possible, an then add extra items of each type to provide for unforeseen control requirements. The guesses at the different possibilities of what the instruments would display are listed below. They would not all be displayed at the same time. The majority of the test-specific instruments would be on the real time data display of the data acquisition system, but if any were needed to fly the MURV during a test, then they would be included in the cockpit design.

- 18 On/Off Indicators

  Examples:

  - Flight mode indicators—approximately 6 flight modes
  - Preprogrammed maneuver indicators—approximately 4 maneuvers
  - Flap position(s)—two-position flaps
  - Speedbrake position(s)—two-position speedbrake(s)
  - Spoiler position(s)
  - Strake positions—two-position strakes
  - Leading-edge slat positions
  - Other two-position control surface positions

- 12 Graduated Indicators

  Examples:

  - Engine RPM
  - Exhaust gas temperature
  - Oil pressure

- Flap position(s)—multiple-position flaps
- Speedbrake position(s)—multiple-position speedbrake(s)
- Strake position(s)—multiple-position strakes
- Other multiple-position control surface positions
- Cylinder head temperature—reciprocating engine
- Sideslip angle
- Machmeter
- G-meter

These indicators were needed to tell the pilot the status of the MURV. The flight mode indicators were needed to let the pilot know which mode the MURV was in at any given time. Similarly, the preprogrammed maneuver indicators were needed to tell the pilot when the computer was in control, executing a certain maneuver. The control surface position indicators were especially needed because in certain flight modes, the computer could move a control surface and the only way a pilot could tell its position would be from the indicator. The last three graduated indicators were also in the *Desired* indicators list. They are included here in case the remote cockpit design finally arrived at does not have all of the desired indicators. Each one of them could conceivably be needed to fly the MURV in one of its configurations or for a specific test.

The number of programmable indicators is a rough estimate, based on a prediction of the maximum number of configuration dependent and test dependent parameters which would need to be displayed at any one time. The number was limited to keep the costs of the data acquisition system and the cockpit lower and to keep the size of the cockpit down so that it could be transported and set up easily. It is possible that the capabilities of the cockpit may be saturated for extremely complex MURV configurations or tests, but enough instruments were provided to make this very unlikely.

8.3.2.6 Summary of Main Features of the Remote Cockpit The remote cockpit would have only those controls and instruments necessary to fly the MURV. Any other displays for real-time data readout would be part of the data acquisition system and the pilot would not use them.

The outside location of the cockpit, with the added feature of the removable handheld control box, would allow the pilot to follow the MURV visually while it was flying in the area of the runway. The cockpit would have a basic set of controls and instruments necessary to fly the MURV regardless of its configuration or the particular test being performed. It would also have "generic" controls and instruments which would be assigned different functions, depending on the MURV configuration and the test it was performing.

A final major feature of the remote cockpit was that it would have a display indicating the position of the MURV in the test area, for the times when it would be flying beyond visual range.

8.4 Design Evaluation

The following will summarize how well the cockpit design met the cockpit objectives described in the beginning of this section. Since the cockpit objectives were derived directly from the MURV overall objectives, the following will also implicitly describe how well the cockpit design met the MURV objectives.

The first objective, to maximize the versatility of the cockpit in terms of the number of different types of tests it could support, was well met by the cockpit design. The control mode select feature would allow different control modes programmed into the flight control computer to be tested. Also, the design allowed the pilot to fly test maneuvers manually or to take advantage of the flight control computer's capability to fly them automatically, for better repeatability of test results.

The cockpit design also did a good job of allowing a large number of MURV aerodynamic configurations to be controllable by the pilot. The real ability to control various configurations was in the flight control system, but the cockpit was designed to insure that the pilot could use the various modes. The cockpit also allowed the pilot to directly control two-position control surfaces such as spoilers and flaps, increasing the number of different configurations which would be controllable by the pilot.

The cost of the cockpit was minimised by designing it to be an outdoor cockpit with a removable handheld control box, instead of having it fully pivoting and/or inside a vehicle with 360 degree visibility. The number of controls and instruments was also restricted to lower the cost.

Restricting the number of controls and instruments and having the cockpit be non-pivoting made the cockpit smaller and lighter. This contributed to the ease of use by making it easier to transport and set up. Reducing the number of instruments and controls also reduced the power requirements of the cockpit (and the flight control and data acquisition systems, as well), thus reducing the size of the generator which might have to be used for power. Ease of use benefitted still more by the fact that restricting the number of items on the cockpit console reduced the pilot workload, making the MURV easier to fly with less training required for the pilot.

The MURV was also made easier to fly by a pilot with previous remote control aircraft piloting experience by having controls similar to those used for current radio controlled aircraft, and also by having a "conventional" flight mode which was identical to standard aircraft three-axis control.

Reducing pilot workload also contributed to the next cockpit objective, which was to minimise risk. With fewer controls to tend to, the pilot could more easily concentrate on his primary task of flying the MURV, and therefore he would be less error-prone. The risk was also greatly reduced by including a display of the MURV location to the pilot, so that he would be less likely to fly beyond control range or out of the designated test area. He would also have an easier time navigating back to the runway for landing, and it would be easier to locate crash sites by knowing the MURV's last detectable position.

Finally, the cockpit's compatibility with the other MURV subsystems was insured by defining the interaction of the cockpit with the ground portions of the data acquisition and flight control systems.

Overall, the cockpit design satisfies the objectives very well. The question that remains is whether an existing cockpit can be procured and modified to incorporate all of these features, or whether the MURV remote cockpit will have to be "custom built".

IX. Data Processing and Recording

The functions of the ground-based components of the data acquisition system are briefly defined in the description given in Section 6.3. The receiver, demodulator and decommutator for the downlink and the transmitter, modulator and commutator for the uplink are described there in more detail and therefore are not covered here. The additional ground-based functional requirements of the data acquisition system are described here and include data processing, data recording, and real-time display.

9.1 Data Processing

In this report, data processing refers to the data manipulation done on the ground after the rf (radio frequency) signal has been received and demodulated, but prior to recording. Processing may also be required for the command inputs to the flight control system. Data processing usually consists of manipulation of the data so that it can be more easily interpreted immediately following the test, or later, during more detailed analysis.

An important function of data processing is the conversion of the raw data into appropriate units. The conversion factor applied to the raw data is a result of prior instrument static calibration and can also be based on other data being collected during the flight. Examples of this would be a parameter measured with a temperature-sensitive transducer that is adjusted based on temperature measurements, or pitot pressure measurements corrected with respect to angle of attack readings. The calibration curves and/or algorithms are stored in the data processing system and used to convert the raw data to engineering units as it is received at the ground station. This calibration process must be done on-line for data being used by the flight control system, pilot station, or real-time displays, but for test-specific data, could be accomplished later, during post-test analysis. It appeared most advantageous to do the calibration of all data in the data processing system. This on-line processing had the advantage that the data could be presented for real-time analysis, or post-test analysis immediately following the test, in a form that was more easily interpreted.

Because the test site will be located some distance from AFIT, it would be desirable to do a preliminary evaluation of test results immediately following the test, so that additional tests could be done and/or the flight test plan modified, before leaving the test site. This might involve the calculation of derived parameters, or parameters whose determination required one or more measured inputs, such as the coefficient of lift. The data processing system should contain the ability to perform these preliminary calculations, as required.

Another data processing requirement was any signal conversion required to interface the ground data system with the pilot station. The instruments and indicators used in the pilot station will most probably require different signal inputs. Some will likely be activated by digital inputs, while for others, the PCM signal will have to be converted to analog.

9.2 Data Recording

The data recording system is a critical element of a flight test system. Most of the analysis of the test results is done after the flight(s) and many times, at another location. Therefore, the success of a flight test depends to a great amount on the accurate recording of the test data. Test data is, for the most part, the parameters measured onboard the aircraft, but may include the pilot commands sent to the flight control system.

Many full-scale flight test aircraft use onboard recording as the primary means of preserving test data. The trend however, tends to be moving towards the use of a telemetry downlink and ground recording, with onboard recording as strictly a backup. In the case of the MURV, there were a number of factors favoring ground recording. First, ground recording was the most feasible solution because much of the data measured onboard was already required on the ground for the pilot, real-time display and/or processing by the flight control system. Additionally, the sub-scale sise of the MURV provided very limiting onboard volume and weight restrictions, the recording system did not have to endure the harsher airborne environmental conditions, and finally, ground recording provided more flexibility in choosing the type of data recording and the form in which the data would be recorded. Providing some onboard recording for relibility reasons would be desirable, subject to availability of onboard space and power.

Post-test analysis can be a time-consuming process, but forethought and planning can help reduce the labor involved in converting the data to a usable form. For these reasons, the objectives for the data recording system were to record the data so that it could be reproduced as accurately as possible, and to provide a method of recording that would reduce the labor involved in post-test data reduction and analysis.

While it was impossible to predict all data analysis techniques that could be used, it was assumed that most post-test analysis would be done using a digital computer. Because of this assumption, and the fact that the telemetered data was already in a digital form (PCM), it was logical to investigate methods of digital recording. Analog recording does offer advantages in some instances, however, and therefore was also examined.

9.2.1 Digital Recording Digital recording was the natural choice for the MURV data system. Its use eliminates the need for signal conversion before recording, or for analysis with digital computers. Additionally, digital recording offers greater error protection than analog recording. The recording/storage mediums considered were magnetic tape and magnetic disc.

Digital magnetic tape is the most widely used method for storing digital flight test data. The use of digital tape with computers is well-developed and has achieved a high degree of standardization [97]. Additionally, direct recording on mainframe computer compatible tape would have significant advantages if mainframe computers were to be used for data reduction.

There are fewer sources of error in digital recording than found in analog tape recording, but digital tape provides much lower packing density. In fact, an important performance measure for digital magnetic tape is the packing density, or the amount of information that can be stored per area of tape. A problem inherent to digital tape is that the probability of drop-out errors increases with increased packing density. Drop-out errors occur when a magnetic tape imperfection causes a loss of a pulse or generation of a spurious signal. Errors in digital data can be much more significant than errors in continuous data, where the values are correlated in time. For this reason, bit error detection codes, such as parity checking, are commonly an intrinsic part of digital recording.

The use of digital recording is limited when high sampling rates are needed. For very high frequency data, such as the example of vibration data, it may be necessary to record on analog magnetic tape. Although IRIG (Inter-Range Instrumentation Group) standards call for different tape recorders for analog and digital recording, development of high-density serial recorders makes it possible to record both on the same recorder [98:1-88].

Another advantage of digital tape recording over analog recording is the ability to randomly access data at different tape positions. Magnetic tape does, however, provide a relatively long access time when compared to a digital disc system.

Data recording on a digital computer disc was another option worth considering. The recording process would be somewhat more complicated for a disc than with tape recording, where the data are recorded sequentially, either as a multiplexed signal or individual parameter signals. Labeling of the data, to include run and data identification information, is required, but can be accomplished during data processing. Disc recording requires a processor to control the allocation of disc space. A computer processing capability is required for data acquisition in any case, therefore the processor could be tasked with the additional job of controlling disc recording.

The decision on the method for digital recording is best delayed until a follow-on study, but should be based on a number of considerations. These include how the data will be used after the test (e.g., mainframe versus personal computer computation), what is the most economical means for permanent storage, and which solution would provide the most flexible, reliable capability for a portable data acquisition system. With the rapid development of microprocessor and personal computer standards, it may be that disc storage will overtake the more common magnetic tape recording for flight test data. This decision is something that should be influenced by the rapid changes in computer capability and application.

9.2.2 Analog Recording The equipment typically used for recording analog signals in flight testing are chart recorders and analog magnetic tape recorders. The two types of chart recorders are the continuous time-recording plotter (strip-chart recorder) and the X-Y plotter. A continuous time-recording plotter moves a length of paper at a constant speed while a pen or a light beam moves vertically across the page, a distance proportional to the magnitude of the incoming signal. An X-Y plotter uses a fixed sheet of paper. The pen moves about the paper and is commanded in the vertical and horizontal directions by two input signals, X and Y.

Chart recorders are a simple economic method for viewing a parameter's variance with time, or the relationship between two parameters, almost immediately after they are measured (there is a lag time associated with every chart recorder). They provide a permanent record of the data, but are more useful for establishing data trends and relationships than for reading actual values. Also, it is time-consuming and inaccurate to convert the traces back into an electric signal.

Analog magnetic tape was considered as a possible storage medium. Its main limitation was the fact that digital data from the receiver would have to be converted to analog data before recording, and then back again to digital for processing. There were also more sources for data error than in digital recording, such as non-linearity, noise, drift and dynamic distortion [11:102]. The major advantage of analog recording over digital recording was the much higher packing density of information on the tape. Additionally, analog recording may be preferred when very high frequency data (greater than 10,000 Hz) is being recorded. Since selection of the specific recording method(s) did not affect the actual MURV design and will affect the ground station only during a more detailed design phase, this decision was delayed until that point.

9.2.3 Video Recording The video camera picture that is transmitted down to the pilot station would most easily be recorded on a high-quality video tape recorder available commercially, with an audio track for recording pilot comments during the flight.

9.3 Real-time Display

Real-time analysis consists of displaying the data in a form that can be used for immediate interpretation of flight data. This might be required during critical phases of a flight where a test engineer would monitor flight conditions during high angle of attack tests, or monitor maximum g-loading during structural tests. Additionally, real-time analysis allows modification to the flight test plan, as needed, to make the most efficient use of flight time.

Data display during the flight or shortly after flight completion could be accomplished by chart recorders, as discussed previously, dial or digital indicators, or using a CRT screen. Display on a computer CRT screen would allow significant flexibility in the displayed information, particularly in the choice of data to be viewed.

9.4 Data Acquisition Ground Station

The ground-based data acquisition system functions of data processing, recording and real-time display should form an integrated system that is flexible to changing experimental requirements. While data processing requirements for the flight control system and pilot station will be somewhat fixed, the test-specific requirements may change many times. The values, if any, that are needed to be displayed during the flight may also vary with each flight. For these reasons, it is important to design a system that is easily modified between flights for the test-specific needs. This flexibility can most easily be achieved by a software-intensive system.

The reliability of the ground station components is critical to system performance. All recording devices, such as the video recorder and the digital recording should be redundant. This is especially important when recording on magnetic tape, where proper data transfer relies on the lack of tape imperfections.

X. Tracking System

Introduction Since the MURV will travel at speeds around 0.3 to 0.4 Mach, it will need to fly beyond visual range in order for it to be able to do any maneuvers besides flying ovals around the runway area. This is inherent in its design as a jet-powered aircraft to be used to perform a wide variety of flight tests. A high-speed vehicle restricted to visual range simply would not be able to perform very many tests.

Safety dictates that the MURV operators (specifically, the pilot) know the location of the MURV at all times when it is out of sight. This is to keep it from straying out of the test area, flying beyond radio control range, or simply from being "lost" and unable to return to the runway and land. Also, if the MURV were to crash, the operators would know its last position and would be able to find the crash site much more easily.

This need created the design for the remote cockpit to include a display which would show the pilot the location of the MURV in real time. To make it easy for the pilot to understand and use for navigation, the display would be an x-y grid, showing the MURV as a moving point or cursor. It would show the MURV location with respect to the ground station and any other landmarks. A desired feature would be its ability to display the range and heading of the MURV in numerical form in addition to the x-y plot.

The cockpit display would need a tracking system to supply it with the MURV location information. Before the design of the tracking system was to be developed, it was first necessary to establish whether the tracking system needed to be part of the MURV system or whether equipment at the test site could be relied upon to provide tracking.

Appendix N contains an investigation into the possibility of using several controlled airspaces within a 150 mile radius of Wright-Patterson AFB. Of these, the U.S. Army Jefferson Proving Ground in southern Indiana is the most likely location for flying the MURV. As indicated in the Appendix, it has a portable radar unit capable of tracking the MURV throughout its flight.

Preliminary discussions with the personnel at Jefferson Proving Ground indicate that they would possibly allow it to be used for MURV flights. If, at a later time, they decide not to allow its use, then this discussion is moot and the MURV design will have to include a tracking system if it is to fly at any of the nearby ranges. The following will assume that the radar at Jefferson Proving Ground can be used, and will discuss the merits of using it versus including a tracking system as a permanent part of the MURV system.

10.1 Objectives

The basis for making a decision in the systems approach is to identify the objectives, weigh the alternatives against those objectives, and select the design which best meets the objectives. Deciding whether to design a MURV-dedicated tracking system is no different. The objectives for this decision were derived from the overall MURV objectives established in Volume One, Chapter II. The first objective, to minimize cost, resulted directly from a MURV objective. The next objective came from the MURV objectives to maximize its versatility in terms of both configurations and tests: to maximize the capability of the MURV to fly at different test sites. For example, if a particular test could not be performed in the confines of the "usual" test area, it would be desirable for the MURV to be able to be transported to a different, larger site, and flown there. Another way of saying this would be to minimize the dependence of the MURV on the equipment at the test site, specifically the tracking equipment.

10.2 Analysis

A MURV design which does not include a tracking system will intially be less expensive than one which does. But if any future testing would require flying at a facility without tracking equipment, the cost of retrofitting a tracking system to the ground system could be much more than including one in the original design. For example, one kind of tracking system uses a rotating dish telemetry antenna to follow the vehicle in flight. Suppose the ground system were originally designed with a simple omnidirectional antenna in the telemetry system because the tracking would be accomplished by radar. The cost of the original antenna plus the cost of replacing it with a pivoting antenna would be more than the cost of designing the system with a pivoting antenna to start with.

A ground system design which includes a tracking system would be more expensive than one without, but the MURV would have the capability to fly at test sites without any tracking equipment.

The questions to be answered are:

1. Will Jefferson Proving Ground be the main test site for the MURV, and if so, will the radar there be available for use?
2. How important is the need to be able to fly the MURV at different locations without tracking equipment?
3. What are the costs associated with different types of tracking systems, including retrofit costs?

Until these questions are answered, it will be impossible to determine whether to include a tracking system into the design or to rely on the radar system at Jefferson Proving Ground.

XI. Airframe Concept Selection

11.1 Approach

Chapter 2.2 described the three airframe concept alternatives for the MURV design:

MURV-1: Delta wing with canard.

MURV-2: Trapezoidal wing with aft tail (conventional).

MURV-3: Delta wing with no tail/canard (tailless).

These configurations were developed in consideration of the three primary design drivers stated in Section 3.1, and according to the performance objectives described in Section 4.4, both found in Volume One. This section presents the decision making process used to select the single airframe concept which was carried forward as the recommended baseline design to be optimized in the next design iteration.

The approach taken was to first review the system needs and objectives in order to identify appropriate measures of effectiveness by which the capabilities of the alternatives could be measured, and assigning weighting factors representing their relative importance in achieving the overall system objectives. With the measures and their weighting factors established, a mathematical relationship, or objective function, was developed which combined all the objective and subjective measures into a single score for each concept. The concept with the highest score was then selected as the baseline configuration which was carried into the next design iteration.

11.2 Selection Criteria

The measures of effectiveness were derived from the objective hierarchy shown in Figure 2.2 of Volume One. For the airframe concept selection, these were separated into two categories: aerodynamics and stability and control. Because all candidates were sized according to the same thrust-to-weight and wing loading ratios (see this volume, Table 2.2), the performance of the three designs was essentially indistinguishable, as is evident from Figure 2.6 and Table 2.8. Therefore, no performance measures were included in the airframe concept selection criteria.

11.2.1 Aerodynamics The two measures used in aerodynamics were $C_{L_{max}}$ and $\alpha_{C_{L_{max}}}$. These were selected to indicate the potential of the vehicle to perform supermaneuvers at extreme attitudes. The values used in the analysis were for a clean aircraft with no high-lift devices and a common wing section and were evaluated using the IDAS program. Obviously, the higher values of both $C_{L_{max}}$ and $\alpha_{C_{L_{max}}}$ indicated greater potential for performing high pitch attitude maneuvers. Also, higher values of $C_{L_{max}}$ allows lower stall speed, thus better landing performance, and higher values for $\alpha_{C_{L_{max}}}$ allows less reliance on the flight control system and additional aerodynamic surfaces to maintain control of the vehicle at higher angles-of-attack.

11.2.2 Stability and Control The designs were evaluated in terms of their ability to perform a variety of experiments and according to their ability to be controllable at extreme maneuver conditions. These measures are denoted as C_{M_α} Control and High-α Control.

11.2.2.1 C_{M_α} Control As mentioned in Volume One, Section 4.5 the quantitative objective measure used in the stability analysis was C_{M_α}. Recall that if $C_{M_\alpha} < 0$ the aircraft is stable, if $C_{M_\alpha} > 0$ it is unstable, and if $C_{M_\alpha} = 0$ it is neutrally stable. Equation 2.9 can be used to calculate C_{M_α} for a conventional wing/horizontal tail configuration, such as MURV-2. That equation is repeated here, together with similar expressions for canard/wing and tailless designs found in Reference [75].

MURV-1:

$$C_{M_\alpha} = C_{L_{\alpha_w}} \frac{X_w}{\bar{c}} + C_{L_{\alpha_c}} \bar{V}_c + C_{M_{\alpha_f}} \tag{11.1}$$

MURV-2:

$$C_{M_\alpha} = C_{L_{\alpha_w}} \frac{X_w}{\bar{c}} - C_{L_{\alpha_t}} (1 - \frac{\partial \epsilon}{\partial \alpha}) \bar{V}_t + C_{M_{\alpha_f}} \tag{11.2}$$

MURV-3:

$$C_{M_\alpha} = C_{L_{\alpha_w}} \frac{X_w}{\bar{c}} + C_{M_{\alpha_i}} \tag{11.3}$$

The inlet contribution to C_{M_α}, $C_{M_{\alpha_i}}$, was assumed to be sero for all three concepts, and $\bar{V}_c$ and $\bar{V}_t$ are calculated as per Equation 4.17 of Volume One, Section 4.5.

Section 2.2 described how $\bar{V}_c$ and $\bar{V}_t$ were estimated, and the baseline values were presented in Table 2.8. These values were calculated for the baseline placement and sizing of the tail and canard. The volumes differ as the placement (moment arm) and size (area) change. Since the sizing and placement of the tail and canard could have easily been altered, a parametric trade study of tail and canard design was performed to identify which configuration allowed the greatest flexibility in achieving a desired level of stability, *i.e.*, value of C_{M_α}.

The procedure was to identify the desired value for C_{M_α}, then solve Equations 11.1 and 11.2 for the values of $\bar{V}_c$ and $\bar{V}_t$ needed to achieve that level of stability. The tail volume was rewritten as

$$l_{t/c} S_{t/c} = \bar{V}_{t/c} S_w \bar{c} \tag{11.4}$$

where the subscript t/c refers to either the tail or canard.

Since the right hand side is constant for a given concept and C_{M_α} value, the range of solutions of $l_{t/c}$ and $S_{t/c}$ which multiply to get that constant value could be found. The design with the smaller product of $l_{t/c} S_{t/c}$ was considered more flexible since, for a given moment arm (placement), the required surface area is smaller; thus, leading to lighter weight and lower hinge moments. MURV-3 could not be evaluated in this way, but was analysed in a more subjective fashion. The analyses are presented in Section 11.4.

11.2.2.2 Hi-α Control Since the MURV is expected to perform unique maneuvers at extreme vehicle attitudes, loss of control power is a vital concern. Control power refers to the ability to generate aerodynamic forces and moments, primarily from the horizontal tail or canard, to change the vehicle attitude in a predictable manner. This was assessed subjectively by examining the configurations for inherent aerodynamic qualities and their potential for sustaining or losing control power at extreme attitudes.

11.3 MOE Weighting Factors

The four measures used in selecting a baseline airframe configuration were: $C_{L_{max}}$, $\alpha_{C_{L_{max}}}$, C_{M_α} Control, and High-α Control. To arrive at weighting values for these measures, they were all initially equally weighted so as to sum to 100; thus, each was initially given a weighting of 25. Deviations were determined for those measures that were more important than the others, or less important.

Of the four measures, High-α Control is the most vital, since it relates not only to the maneuverability of the vehicle, but to the safety of operation as well. Therefore, the weighting value for this measure was increased to 35.

Conversely, $C_{L_{max}}$ was considered least important of the four, since the value estimated did not include the effects of high-lift devices. The values of $C_{L_{max}}$ provided insight into the degree of additional lift needed from such devices, so that the design with the highest $C_{L_{max}}$ value would not require as complex of a high-lift device design as that of the others. Yet all three designs could be optimized with high-lift devices such as flaps or leading edge slats, and the values of $C_{L_{max}}$ increased dramatically. Thus, the weighting factor for $C_{L_{max}}$ was reduced to 15.

The last two measures, $\alpha_{C_{L_{max}}}$ and C_{M_α} Control, were considered to be of equal importance since both give an indication of the aerodynamic efficiency of the concept. $\alpha_{C_{L_{max}}}$ indicates the ability to pitch the aircraft to an extreme attitude before the wing stalls, or loses significant lift. C_{M_α} Control indicates the ability to alter the pitching moment stability by a relatively simple relocation of the canard or horizontal tail. Both measures are important to the success of the MURV as a modular vehicle attempting high angle-of-attack maneuvers, and are therefore equally weighted at 25.

11.4 Evaluation

The estimated values for all the measures of effectiveness are shown in Table 11.2 and are labeled as "values". The corresponding "score" is a normalised value, where the maximum of the three candidates received a score of unity. The other scores were computed as the MOE value divided by the maximum, with the exception of the normalized score for C_{M_α} Control, which is explained later. The total score for each design was found by multiplying the measure weighting factor by the score for each objective measure, and summing. An ideal design would have a total score of 100.

The aerodynamic evaluation involved a straight-forward compilation of the two measures, $C_{L_{max}}$ and $\alpha_{C_{L_{max}}}$. Note that MURV-1 had the highest $C_{L_{max}}$ value, yet the lowest $\alpha_{C_{L_{max}}}$, while MURV-2 had the highest $\alpha_{C_{L_{max}}}$ estimate. After normalization, MURV-2 had slightly better scores than MURV-1, and both exceeded MURV-3.

The evaluation of the stability and control characteristics was not so simple, however. The objective measure was the flexibility in achieving a neutrally stable design ($C_{M_\alpha} = 0$) which, for MURV-1 and MURV-2, was assessed as the canard volume and horizontal tail volume required for a given wing and c.g. location, where a lower tail/canard volume is preferred. The label given to this measure is C_{M_α} Control. The values in Table 11.2 are the required volumes to achieve $C_{M_\alpha} = 0$, and the corresponding scores are normalised differently since, for this measure, lower is better. The normalization of the score for candidate i was found as

$$\text{Score}_i = 1 - \frac{\bar{V}_i - \bar{V}_{low}}{\bar{V}_{low}}$$

where $\bar{V}_i$ is the tail/canard volume for candidate i, and $\bar{V}_{low}$ is the lower tail volume value. Note that when $\bar{V}_i$ equals $\bar{V}_{low}$, Score_i equals one. The values in the table were computed by solving

Equations 11.1 and 11.2 for the volume values needed for $C_{M_\alpha} = 0$ with all other parameters known.

Quite obviously, this approach could not be used for MURV-3. To arrive at a value and score for MURV-3, a more subjective assessment of flexibility was made by comparing the expressions for C_{M_α} for the three concepts, i.e., Equations 11.1, 11.2, and 11.3.

Note that the first right hand term in Equation 11.3 is the same as the first term in Equations 11.1 and 11.2. Once the wing section and high lift devices are chosen, the only controllable parameter in altering C_{M_α} for MURV-3 is X_w, which is the horizontal distance from the wing aerodynamic center to the vehicle center of gravity. (Recall that for all three candidates, $C_{M_{\alpha_i}}$ was assumed to be zero.) Therefore, the only means of controlling C_{M_α} is by moving either the wing or the c.g. This same flexibility exists for MURV-1 and MURV-2, however. Conversely, both MURV-1 and MURV-2 have additional freedom in setting C_{M_α} due to the existence of an additional control surface (canard and horizontal tail, respectively). The ability to assume various levels of stability is an important design feature for the MURV. A baseline configuration which allows such a variation by a rather simple relocation of the canard or tail, as opposed to one which requires that a new surface be *added*, is an important advantage. Due to its lack of flexibility in this regard, MURV-3 was given a value and score of zero for the C_{M_α} Control measure.

Stability and control was assessed in one other subjective area as well. A particularly important aspect was evaluating the ability to control the vehicle at extreme maneuver conditions, such as high angles-of-attack. MURV-2 had the disadvantage of aerodynamic "blanking" of the horizontal tail at high α conditions, as discussed previously. MURV-3 had no additional control surface to affect, yet this also meant that it had only one means of generating all aerodynamic moments at all flight conditions. This requires a complex arrangement of deployable surfaces, all on the wing leading and trailing edges, to generate the moments needed to maneuver the vehicle. MURV-1 did not have the problem of aerodynamic interference from the wing wake, and the canards can be used to generate positive or negative pitching moments through a wide range of flight conditions by deflecting them to the proper incidence angle. For these reasons, MURV-1 was considered to have the overall advantage at extreme maneuver conditions, followed by MURV-2, then MURV-3.

To quantify these evaluations a matrix of preferences was assembled, called a preference structure, which contained one-to-one comparisons of each candidate versus the other two for their controllability at high angles-of-attack. The level of preference of one design over another was assigned according to the following scale

$$\begin{aligned} 1 &= \text{equivalent} \\ 3 &= \text{slightly preferable} \\ 5 &= \text{preferable} \\ 7 &= \text{more preferable} \\ 9 &= \text{highly preferable} \end{aligned}$$

If one design was considered as *more preferable* over another, it received a value of 7 in the preference matrix comparison against that design. This also meant that the second design was one-seventh as preferable as the first. When all possible comparisons of candidate i versus candidate j were assigned a preference value, v_{ij}, candidate i received a composite preferece value, V_i

$$V_i = \left[\prod_{j=1}^{3} v_{ij}\right]^{\frac{1}{3}} \tag{11.5}$$

Table 11.1 shows the resulting preference structure for the three candidate designs. The table can be read as follows: "(row candidate) is (preference value meaning) to (column candidate)." For example, MURV-2 is slightly preferable to MURV-3, since $v_{23} = 3$.

As before, the scores in Table 11.2 reflect normalizing all values so that the maximum is unity.

Table 11.2 shows the MOE values and scores for the three MURV designs. After applying the weighting factors to the normalized scores for each measure, MURV-1 came out as the preferred baseline design, with a total score of 97.83 out of a possible 100.

Table 11.1. MURV High-α Control Preference Structure

	MURV-1	MURV-2	MURV-3	V_i
MURV-1	1	5	7	3.271
MURV-2	1/5	1	3	0.840
MURV-3	1/7	1/3	1	0.363

Table 11.2. MURV Airframe Concept Evaluation

Category	MOE	Wtg Fac	MURV-1		MURV-2		MURV-3	
			Value	Score	Value	Score	Value	Score
Aerodynamics	$C_{L_{max}}$	15.0	1.12	1.0	1.05	0.938	0.99	0.884
	$\alpha_{C_{L_{max}}}$	25.0	24.4°	0.913	26.7°	1.0	25.2°	0.944
Stability and Control	C_{M_α} Control	25.0	18.16	1.0	22.7	0.75	0.00	0.0
	Hi-α Control	35.0	3.271	1.0	0.84	0.257	0.363	0.111
	SCORE	100.0		97.83		66.82		40.75

11.5 The Preferred Airframe Concept

The preferred baseline configuration for the MURV is a delta wing and canard configuration. MURV-1 was clearly superior in the comparisons made in the measures of aerodynamics and stability and control, and was equivalent to the other candidate designs in performance characteristics. Its ability to perform high-α maneuvers without risk of losing control power gave it a significant advantage over MURV-3, and the flexibility in achieving a neutrally stable design demonstrated its superiority over MURV-2. In fact, in all measures except one, MURV-1 was the superior design. Therefore, this design was carried into the next iteration of the Preliminary Design Phase — Airframe/Engine Evaluation and Configuration Optimization. In this design iteration a trade study of the two most promising propulsion systems, as determined by the propulsion system evaluation, was performed to assist in selecting the single engine/airframe configuraiton as the recommended baseline design.

11.6 Airframe/Engine Evaluation

The propulsion system assumed for the initial development of all the MURV concepts consisted of two Microdynamics prototype engines mounted side-by-side. These, along with other candidate engine designs, were rejected by the propulsion system evaluation in favor of the Teledyne 312 and 320 engines (see this volume, Chapter III). An airframe/engine performance trade study was conducted to provide information for selecting a single engine design for the baseline MURV. The configuration of MURV-1 was then modified to accommodate the favored engine.

11.6.1 Configuration Optimization To avoid favoring one engine over another, a derivative MURV-1 configuration was optimised for each engine. The airframe/engine configuration which had the greater value of the measure of effectiveness was selected as the baseline design. This was accomplished by generating a nonlinear optimisation problem which maximised the selected measure of effectiveness, while satisfying the pertinent design and performance constraints. The optimal design was expressed as a set of geometric and performance sizing parameters. The sizing and geometric parameters of interest were T/W, W/S, AR, Λ_{LE}, TOGW, and the amount of fuel required, W_F.

The primary objective was to determine the combination of these design parameters which yielded the most optimum performance capabilities at a supermaneuver flight condition. This particular objective was selected for two reasons: (1) Herbst recommended that a vehicle ought to have a thrust-to-weight of near 1.0 to attempt a post-stall turn, and (2) this condition places the most severe constraints on the design in terms of maximizing thrust and minimizing weight and drag. The approach taken was to maximize an appropriate performance measure at the supermaneuver condition subject to all the conventional design constraints plus other performance requirements, such as meeting the takeoff and landing distance requirements.

To solve the maximization problem the General INteracive Optimizer computer program (GINO) was used. Appendix M describes the usage of the GINO program and also lists the specific problem formulation which resulted here.

11.6.1.1 Optimization Objective As stated above, the purpose was to optimise the airframe/engine combination capabilities at a supermaneuver condition. T/W appeared to be the ideal parameter to optimise, since a required range for it is mentioned specifically by Herbst. However, the potential maneuver capability is more completely assessed by computing the specific excess power, P_s

$$P_s = \frac{(T\cos\alpha - D)V}{W} \tag{11.6}$$

where T, D, W, and α are oriented as in Figure 2.5. The specific excess power involves not only the benefits of T/W, but the penalty for drag and the influence of velocity and weight, and so represents a more complete measure of the vehicle's maneuvering potential. Therefore, to optimise the MURV's capability to perform supermaneuvers, the objective for this study was to maximise P_s at a typical supermaneuver flight condition. The anticipated flight conditions for executing a post-stall turn were described in Volume One, Section 4.2. For this analysis, the flight condition selected was Mach 0.2 at 6000 feet. At this Mach and altitude, the velocity is 219 feet per second (fps), or about 130 knots true airspeed and 119 knots equivalent airspeed. The maximisation problem can be expressed as

Maximize:

$$P_s = \frac{(T\cos\alpha - D)V}{W}$$

Subject to:

$$C_i \qquad i = 1,\ldots,k$$

$$C_j \qquad j = k+1,\ldots,k+m$$

where there were k design constraints and m performance constraints. Each constraint type is explained in the following sections.

11.6.1.2 Design Constraints The k design constraints were a result of either geometric limitations or subjective limitations based on the MURV design requirements. The design requirement of a fighter-like configuration placed limits on the wing aspect ratio and the leading edge sweep angle. Section 2.2 described the range of aspect ratios for typical modern fighter designs, and from this range an upper limit of 3.5 was used. From a similar data base also described in Section 2.2, the leading edge sweep angle was constrained to be greater than 45°. These values represented the limits of most typical fighter aircraft wing designs. The problem formulation which follows does not address the effect of Λ_{LE} on performance, though it surely has an effect. In general, however, the less sweep back there is to the wing, the more efficient it becomes. We knew *a priori* that the optimum solution for Λ_{LE} would be at the forward most sweep angle. Therefore, Λ_{LE} was not specifically included in the setup of the problem.

The wing span was also limited to avoid transportability problems to and from the test site. In order to fit the vehicle onto a trailer and carry it over public highways, an upper limit of 10 feet was established for the wing span. This limit assumed that the wing would be attached to the airframe before transporting it to the test site.

The physical limitations on the design resulted from forcing the solutions to represent feasible designs, particularly in terms of the estimated weights. In order for a solution to be feasible, the estimated weight had to be consistent with the method of estimating TOGW described in Section 2.1.1; that is, the computed weight value must be representative of a feasible design. Without such a constraint, the best solution would be to have zero weight, thus an infinite value for T/W. To obtain a reasonable weight estimate, the TOGW was calculated two ways in the program: (1) as the ratio of the thrust divided by T/W, and (2) using a simplified estimate of TOGW based on the method of Section 2.1.1. By forcing the two weight estimates to converge, a feasible solution was made possible.

Other physical constraints which had to be satisfied were the mathematical relationships for lift and drag at the flight conditions of interest. From Equation 11.6, the aerodynamic drag, D, had to be calculated in order to determine P_s. The assumption of a quadratic relationship between lift coefficient, C_L and drag coefficient, C_D, is a classical one

$$C_D = C_{D_0} + kC_L^2 \tag{11.7}$$

where $k = (\pi ARe)^{-1}$, and e is the *span efficiency factor*, usually determined by experiment. For MURV-1, e was found from drag data provided by the IDAS program to be 0.565 and, from the same data at Mach 0.2 and 6000 feet, $C_{D_0} = 0.025$. Since aspect ratio was a variable parameter, drag changed inversely proportional to it and thereby affected the computed P_s values.

11.6.1.3 Performance Constraints The performance constraints forced any feasible solution to meet the takeoff and landing constraints as described in Voume One, Section 4.4. A functional relationship of the design parameters and the takeoff and landing constraint was required for use in the optimization program. The form of the takeoff constraint was derived empirically from determining the range of design parameters which allowed the takeoff distance constraint to be met. To determine this range, a computer program was written which estimated the takeoff distance required for a given engine thrust, drag coefficient, wing loading, and $C_{L_{max}}$ by calculating an average acceleration due to these forces over a range of velocities. This average acceleration was then applied to the vehicle until takeoff speed was reached. The takeoff speed was assumed to be 20% above stall speed, where stall speed is

$$V_{st} = \sqrt{\frac{2W/S}{\rho C_{L_{max}}}} \tag{11.8}$$

Velocity was computed at discrete time increments, and the ground roll distance was summed over all increments until takeoff speed was reached. T/W and W/S were varied systematically to find the combinations which allowed the takeoff distance constraint to be met; i.e., reach takeoff speed in less than 480 feet. Figure 11.1 shows the resulting relationship, simplified as a linear function, between minimum allowable thrust-to-weight and wing loading at takeoff.

The inequality constraint line is defined by

$$(T/W)_{TO} \geq 0.07 + 0.04(W/S)_{TO} \tag{11.9}$$

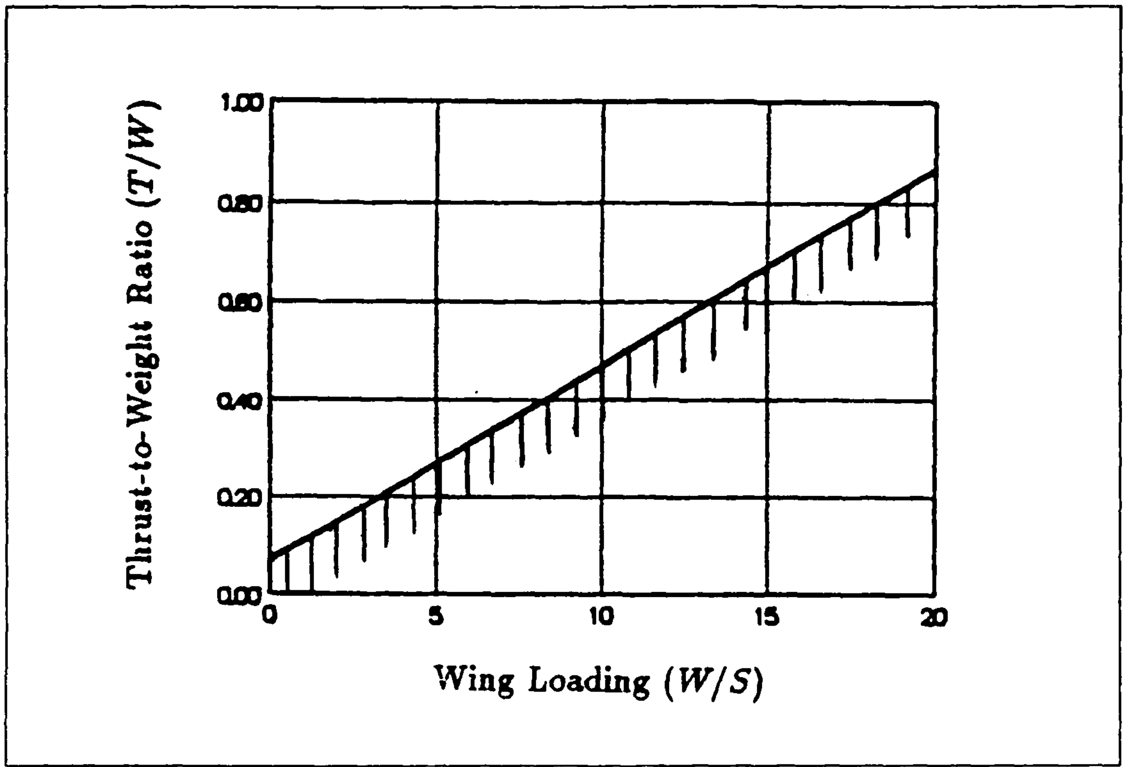


Figure 11.1. Minimum Allowable T/W at Takeoff versus Wing Loading

where the subscript TO implies conditions at takeoff. Equation 11.9 represents the performance requirement for the minimum allowable T/W to achieve the takeoff distance as a function of wing loading. In similar fashion, a constraint of maximum allowable wing loading versus $C_{L_{max}}$ at takeoff was found for a nominal T/W value of 0.5; a higher thrust-to-weight would allow a higher wing loading to meet the constraint. The resulting inequality constraint was

$$(W/S)_{TO} \leq 8.8 C_{L_{max}} \tag{11.10}$$

These equations ensured that any feasible solution would meet the takeoff requirements for the 600 foot runway at Jefferson Proving Grounds, Indiana.

The functional form for the landing constraint came from Equation 4.15 in Volume One, Section 4.4. Substituting the known quantities for the takeoff ground roll (480 feet), and air density at the JPG ground level of 1000 feet (.00231 slugs/cu.ft.), and assuming C_D = 0.2 and

$\mu_h = 0.55$

$$(W/S)_L \leq \frac{13.6257C_{L_{max}} - 7.1330}{\ln(1.9097C_{L_{max}})} \quad (11.11)$$

In computing a value of weight at landing, a worst case scenario was assumed where the aircraft had to immediately land after takeoff and only five pounds of fuel was burned, or $W_L = TOGW - 5$.

Another performance constraint was the calculation of the amount of thrust and weight at the start of the maneuver. Based on the evaluation of the propulsion system for the Teledyne engines, the thrust at Mach 0.2 and 6000 feet is approximately 82% of the value at sea level, Mach 0. The weight differs from TOGW only in the amount of fuel assumed to remain in the tank at the supermaneuver flight condition. The T/W value at this condition strongly depends on the amount of fuel remaining, since the weight is reduced by the amount of fuel burned. The relationship for T/W at the maneuver condition is

$$(T/W)_M = \frac{T_M}{W_M} = \frac{.82T_{SLS}}{TOGW - W_F(1 - p_M)} \quad (11.12)$$

where W_F is the total amount of fuel, p_M is the percentage of fuel remaining in the fuel tank at the start of the maneuver, and the subscript SLS refers to conditions at sea level, static (Mach=0.0). W_F is dictated by the engine specific fuel consumption, or SFC, the engine thrust, T_{SLS}, and the time of flight requirement for full power, t_f. SFC is defined as the amount of fuel required to deliver one pound of thrust, and is equal to the fuel flow rate divided by the engine thrust. In the previous analyses of Chapter 2.1.1, t_f was assumed to be thirty minutes for all weight studies. During the optimization process, t_f was varied to determine the sensivity of the vehicle weight and performance to various fuel tank sizes. The total amount of fuel was calculated by

$$W_F = T_{SLS}\mathrm{SFC}\frac{t_f}{60}$$

where t_f is in minutes and SFC has units of $\frac{lb_{fuel}/hr}{lb_{thrust}}$.

11.6.1.4 Optimization Problem With the objective function and all constraints identified, the complete optimization problem was solved for a given engine type, which fixed the values of T_{SLS} and SFC. Assuming $\alpha = 0°$, the problem was expressed as

Maximise:

$$P_{S_M} = \frac{219(T_M - D_M)}{W_M}$$

Subject to: (design inequality and equality constraints)

$$\begin{aligned}
AR &< 3.5 \\
b &< 10 \\
W_F &= T_{SLS}\mathrm{SFC}\frac{t_f}{60} \\
W_E &= 0.521W \\
W_1 &= \frac{T_{SLS}}{(T/W)_{TO}} \\
W_2 &= W_E + W_{PL} + W_F \\
W &= 1/2(W_1 + W_2) \\
|\Delta W| &< 0.01 \\
\Delta W &= 1 - W_1/W_2 \\
(T/W)_{TO} &= T_{SLS}/W \\
T_M &= 0.82T_{SLS} \\
W_M &= W - W_F(1 - p_M) \\
D_M &= C_{D_M} q_M S \\
C_{L_M} &= \frac{W_M}{q_M S} \\
C_{D_M} &= 0.025 + \frac{C_{L_M}^2}{0.565\pi AR} \\
(T/W)_M &= T_M/W_M
\end{aligned}$$

(performance inequality constraints)

$$(T/W)_{TO} \geq 0.07 + 0.04(W/S)_{TO}$$

$$(W/S)_{TO} \leq 8.8 C_{L_{max}}$$

$$(W/S)_L \leq \frac{13.6257 C_{L_{max}} - 7.133}{\ln(1.9097 C_{L_{max}})}$$

11.6.2 Analysis The independent variables in this formulation were T_{SLS}, SFC, p_M, t_f, W_{PL}, and $C_{L_{max}}$. T_{SLS} and SFC were known for bo'h the Teledyne 312 and 320 engines. Trade studies were performed to isolate differences in vehicle maneuvering capabilities between the two candidate engines. The sensitivities investigated were the fuel tank sizing and its effect on the vehicle gross weight and thrust-to-weight, and the amount of fuel remaining in the tank at the start of the maneuver and its effect on thrust-to-weight at the maneuver condition.

11.6.2.1 TOGW Sensitivity to W_F Section 2.1.1 presented the buildup of the gross weight estimate for the MURV concepts. There it was stated that the fuel tank was sized for continuous thirty minutes of operation at sea level at maximum power. It was noted that this was a conservative estimate for the amount of fuel required for a typical experimental mission. To arrive at a more reasonable and optimal design, the fuel tank size was varied for 20, 25, and 30 minutes of operation at maximum power at sea level, and the variation in TOGW was observed for both the Teledyne 312 and 320 engines. Feasible design solutions were found using the optimization problem formulation while varying the fuel loading parameter t_f (t_f = minutes of fuel available). Figure 11.2 shows the sensitivity that resulted.

All other design parameters, such as *AR* and W/S, were set at their optimum levels as determined by the optimal value of maneuver P_s calculated. That is, as t_f was varied, the optimization problem gave the solution which maximized the energy maneuverability, P_s, at the maneuver condition. The values of all design parameters were free to migrate to their optimal settings, with the exception of t_f and those that were fixed for a given engine. Note that for all values of t_f the gross

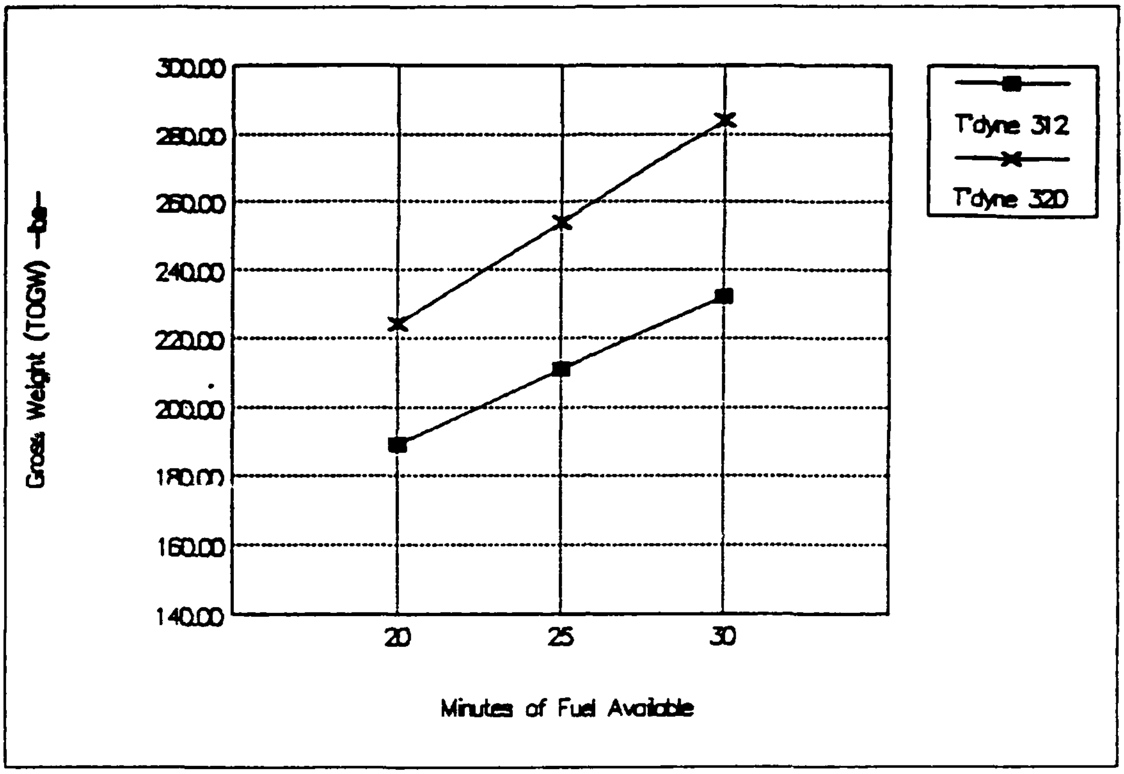


Figure 11.2. TOGW Sensitivity to Fuel Load - t_f

weight of the Teledyne 320-powered vehicle exceeds that of the 312-powered MURV, the difference ranging from about 35 pounds to 50 pounds. For the Teledyne 312-powered vehicle, the difference in TOGW for increasing the fuel loading from 20 minutes to 30 minutes is about 43 pounds, while for the Teledyne 320, an additional 60 pounds is required.

11.6.2.2 Maneuver T/W Sensitivity to p_M This trade study determined the sensitivity of maximum achievable T/W values at the supermaneuver flight condition as a function of the percentage of fuel remaining in the fuel tank at the start of the maneuver. For this trade study, the fuel tank capacity was varied from 20 minutes of fuel to 30 minutes, for both candidate engines. Figure 11.3 shows the results of the trade study, where the T/W value is the maximum achievable within all the constraints identified, and all geometric sizing parameters are set to their optimum levels as determined by the GINO solution. The graph on the left is for the Teledyne 312, and that on the right is for the Teledyne 320. The Teledyne 320-powered MURV achieves much greater T/W levels than with the Teledyne 312 at all equivalent conditions, thus there is a significant advantage in maneuverability with the Teledyne 320. However, as was discovered in the previous trade study, the gross weight of the MURV with the Teledyne 320 is about 35 to 50 pounds heavier, depending on the total amount of fuel required.

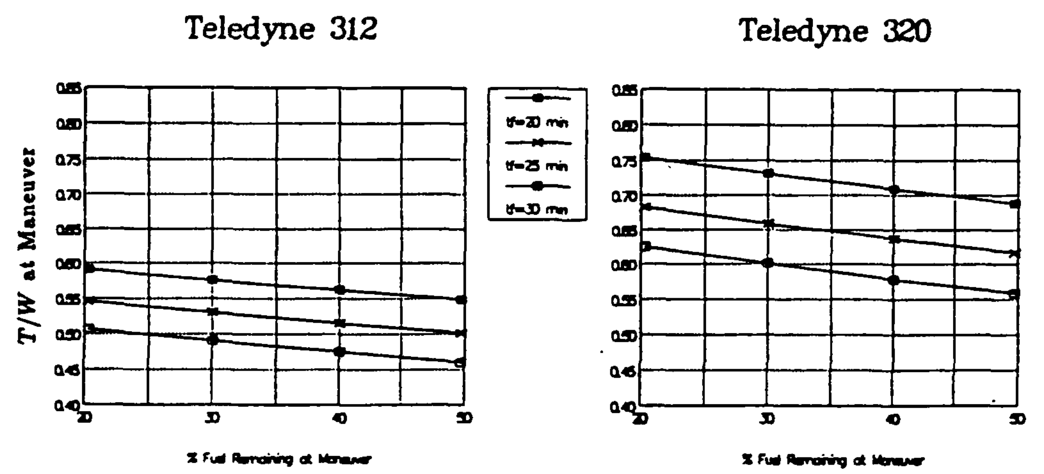


Figure 11.3. Maneuver T/W Sensitivity to t_f and p_M

The percentage increase in maneuver T/W of the Teledyne 320-powered MURV over the Teledyne 312-powered vehicle ranges from 21.3% to 27.6%, depending on the combination of t_f and p_M. Most importantly, there are no conditions for which the 312-powered MURV can achieve a maneuver T/W greater than 0.6. On the other hand, the 320-powered vehicle can achieve a maneuver T/W greater than 0.7 for several combinations of t_f and p_M, and meets or exceeds 0.6 for the majority of fuel loading conditions.

11.6.2.3 Recommended Propulsion System The engine selection depended on the relative importance of minimizing TOGW versus maximizing P_s at the maneuver condition. By minimizing weight, the vehicle is easier to transport and handle on the ground. It would most likely be cheaper to manufacture and operate, since the lighter weight version had the Teledyne 312 engine, which burns less fuel than the other candidate engine. However, the 320-powered MURV has much greater maneuver capability, which is vital to performing the primary mission — supermaneuverability experiments, such as post-stall turns and maneuvers. These class of experiments require high T/W values, well above the projected capabilities of the 312-powered design. The Teledyne 320 allows greater capacity for weight growth since, for all other experiments considered in this study, it is somewhat overpowered. This advantage offset the fact that several of the subsystems and their associated weights were yet undefined.

The value system dilemma was resolved after recognizing that with the Teledyne 312, one of the three primary design drivers was not adequately fulfilled — the ability to conduct the *unique* class of experiments in supermaneuverablity. Without this capability, the MURV concept, as formulated from the very beginning, was not complete. The risk of weight growth further supported the Teledyne 320, since it has the greater excess thrust capability. The weight of the 320-powered MURV would have to increase about 20%, or about 50 pounds, before its T/W is reduced to the same level as the 312-powered vehicle. This might be viewed as doubling the available payload capability, whereas no such margin exists for the Teledyne 312. As a result of these comparisons, the recommended engine for use in the baseline MURV design is the Teledyne 320.

11.6.2.4 Fuel Loading Trade Study With the propulsion system characteristics decided, the optimal fuel loading, t_f, had yet to be determined. Up to this point the fuel tank had always been sized for 30 minutes at maximum sea level power. A trade study was conducted to determine the optimal fuel loading which maximized the thrust-to-weight at the supermaneuver condition, yet still allowed for sufficient flight time for performing experiments. Table 11.3 shows the effect of t_f on the solutions of the optimal values for TOGW, AR, wing area S, wing span b, and fuel tank capacity W_F.

Table 11.3. MURV-320 Sizing Parameters for Various t_f Values

t_f (min)	TOGW (lb)	AR (-)	S (ft^2)	b (ft)	W_F (lb)
20	224	3.5	19.91	8.35	57.4
25	254	3.5	22.57	8.89	71.8
30	284	3.5	25.22	9.40	86.1

Figure 11.3 showed the sensitivity of maneuver T/W to fuel loading for various percentages of fuel remaining at the start of the maneuver. The information in Table 11.3 and Figure 11.3 suggests that, for a goal maneuver T/W of 0.7, the maximum fuel loading allowable is about 25 minutes of fuel (72 pounds), and the maneuver should not be attempted with more than about 25 to 30 percent of fuel remaining in the tank. If, following the maneuver, the thrust is reduced to one-third power, for the same specific fuel consumption rate (SFC=1.05) there would be enough fuel for an additional 19 to 23 minutes of flight time. This assumes that the maneuver itself uses a negligible amount of fuel. Considering that a Herbst maneuver would likely be sustained for only about 5 seconds, which translates into about 15 seconds of fuel at one-third power (see Volume One, Section 4.4.1), this was not a particularly limiting assumption.

Given these observations, the fuel loading recommended is to support 25 minutes of operation at maximum power at sea level, or a fuel tank containing about 72 pounds of fuel (11.08 gallons).

11.7 Summary

For both airframe/engine designs, the optimal values for AR and $C_{L_{max}}$ were obtained at their upper limits. This was not too surprising, though we anticipated a stronger influence of the landing constraints on possible values of $C_{L_{max}}$. The only term where AR appears is in the induced drag term for C_D, so that as AR increases the induced drag decreases, and overall drag decreases. Thus, increasing AR leads to increased P_s values.

The recommended baseline design is a delta wing and canard configuration installed with a Teledyne 320 propulsion system, with enough fuel to operate continuously at maximum power at sea level for 25 minutes. The aspect ratio of the optimised MURV-1 configuration was increased to 3.5 and the goal value for $C_{L_{max}}$ was set to 1.6, though a higher value is desirable if no overriding penalties are incurred. Although Λ_{LE} was not determined from these trade studies, it was advantageous to reduce the leading edge sweep angle to the minimum acceptable value of 45° to improve the aerodynamic efficiency of the wing design. This reduces the induced drag contribution at high angles-of-attack, and is also more compatible with a higher aspect ratio planform.

This configuration was carried forward into the final iteration of the preliminary design development for the external configuration. The MURV-1 installed with the Teledyne 320 combines the airframe with the greatest aerodynamic flexibility with the engine which produces the most maneuverable vehicle. The resulting design is one which is consistent with the maneuvering T/W recommendation of Herbst for performing post-stall maneuvers, thus fulfilling the requirement of one of the three primary design drivers of the MURV concept.

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Block 18 Remotely Piloted Vehicles
Unmanned
Flight Testing
Modular Construction
Flight Control Systems
Digital Systems
Digital Communications
Distributed Data Processing
Data Acquisition
Landing Gear
Turbojet Engines

Block 19 This thesis presents the analysis and development of a modular unmanned research vehicle (MURV) to support aeronautical research for the Air Force Institute of Technology. The MURV is proposed as a test vehicle to permit experimental efforts beyond the restrictions of pure analytical and wind tunnel research, yet less costly and more accessible than full-scale flight tests. A classical systems approach was applied, in concert with a conventional aircraft design process, which emphasized system level needs and objectives in the design of MURV subsystems. The primary design drivers were the need for adequate data aquisition for anticipated experiments, structural and functional modularity to permit simple reconfiguration, and focus on a set of unique experiments relating to fighter-like supermaneuverability. The supermaneuverability experiments dictated that the general arrangement of the MURV baseline design would resemble a typical modern fighter aircraft configuration, the recommended baseline being a turbojet-powered delta wing design with canards, single vertical tail, and control-configured ventral fins. Modularity implications resulted in the design of a flexible, digital flight control system with primary functions distributed between the vehicle and a remote pilot/control ground station and a fuselge design which allows for relocation and replacement of wings and tails or canards. The data acquisition system is fully integrated with the flight control system and the remote ground station. The MURV is capable of flight speeds approaching 260 knots for altitudes up to 20,000 feet, and has fuel to fly for well over 30 minutes. Several follow-on studies are identified which are necessary to complete the design and bring the MURV to an operational status.